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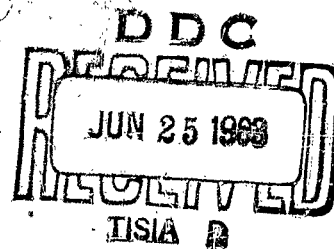
STUDY OF SPACE MAINTENANCE TECHNIQUES

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May 1963

Directorate of Aeromechanics
Aeronautical Systems Division
Air Force Systems Command
Wright-Patterson Air Force Base, Ohio

Project No. 8170, Task No. 817008



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by Bell Aerosystems Company, Aerospace/Rockets Division, Buffalo, N.Y.;
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FOREWORD

This study was initiated by the Support Techniques Branch of the Flight Accessories Laboratory, Aeronautical Systems Division, Air Force Systems Command, Wright-Patterson Air Force Base, Ohio. Mr. Peter Van Schaik, Logistics, Maintenance and Support Techniques Section, Support Techniques Branch, Flight Accessories Laboratory was the contract monitor.

The study was conducted by the Bell Aerosystems Company, of the Bell Aerospace Corporation, Buffalo, New York, under Contract No. AF 33(657)-7838. Dr. L. M. Seale, Chief, Space Systems, Advanced Systems Design, was the principle investigator for the Bell Aerosystems Company. The work performed was in support of Project No. 8170 and Task No. 817008. The research sponsored by this contract was initiated in January 1962 and was completed in November 1962.

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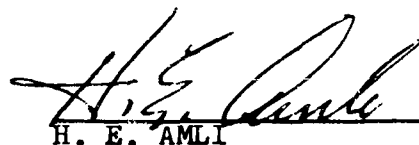
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ABSTRACT

This report describes the Study of Space Maintenance Techniques. The study was devoted to establishing preliminary concepts for: maintenance techniques, assembly techniques for space stations, repair techniques for system components, applicability of terrestrial tools and fasteners, and criteria for remote manipulators. To accomplish these ends, five manned space systems were studied to identify the operating environments and the maintenance, assembly, and repair functions. At the conclusion of these system requirements studies, the capability of the man to accomplish the maintenance activities inside the vehicle, and exterior to it, was estimated while the man functioned in a "shirt-sleeve" environment in a space suit and in a small maintenance capsule. Simple experimentation, on an air-bearing platform was conducted to establish maintenance task times and the feasibility of various fasteners and tools for space maintenance.

This report has been reviewed and is approved.



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CONTENTS

Chapter		Page
I	INTRODUCTION	1
	A. Objectives	1
	B. Approach	1
	C. Study Limitations	4
	D. Recommendations	5
2	SYSTEM DESCRIPTIONS	6
	A. Permanent Multimanned Space Stations	7
	1. Earth Satellite Weapons Systems (ESWS)	7
	2. Military Test Space Station (MTSS)	7
	B. Space Station Supply Vehicle	11
	C. Lunar Landing Vehicle	14
	D. Orbital Maintenance Operational Systems	14
	1. Space Maintenance Capsules	15
	2. Extravehicular Space Suit	17
3	SYSTEM IDENTIFICATION AND MAINTENANCE REQUIREMENTS	18
	A. General	18
	B. Terminal Rendezvous, Retrieval and Docking	22
	1. Mechanical Coupling Techniques	22
	2. Ejectable Missile Techniques	24
	3. Projectable Arm Devices	26
	4. Direct Manned Participation Methods	26
	5. Optical Rendezvous and Docking Methods	26
	6. Maintenance Requirements	26
	C. Space Station Assembly	31
	1. Assembly-in-Orbit Techniques	32
	2. Self-Erecting Space Stations	32
	3. Convertible Space Stations	33
	4. Inflatable Space Stations	33
	D. Orbital Assembly Fastening and Sealing Techniques	33
	1. Pyrotechnic Devices	33
	2. Module Seals	33
	E. Space Vehicle Structures	34
	1. Structural Configurations	34
	2. Maintenance Analysis - Space Vehicle Structure	39
	F. Space Vehicle Propulsion	42
	1. Propulsion System Descriptions	44
	2. Maintenance Requirements	52

CONTENTS (CONT)

Chapter		Page
	G. Life Support Systems	63
	1. System Concepts	63
	2. Maintenance Analysis	67
	H. Space Vehicle Power Supplies	72
	1. Nuclear-Turboelectric	73
	2. Piston Engine	73
	3. Photovoltaic Cells	73
	4. Chemical Turboelectric	74
	5. Fuel Cell	74
	6. Dynamic Heat Engine	75
	7. Maintenance Analysis	75
	I. Avionics Systems	78
	1. Rendezvous Guidance	79
	2. Communications	79
	3. Attitude Control and Stabilization	83
	4. Maintenance Analysis - Avionics Systems	83
4	MAINTENANCE TASK ANALYSES	87
5	EFFECTS OF SPACE ENVIRONMENTS ON SPACE MAINTEN- ANCE	100
	A. General	100
	B. Meteoroids and Their Effects on Space Structures	101
	1. Particle Characteristics	102
	2. Meteor Flux Density and Mass	102
	3. Impact Probability	103
	4. Impact Phenomena	103
	5. Penetration and Spalling Probability Calculations	108
	6. Effects After Impact	118
	7. Self-Sealing Repair Methods	130
	C. Particle Radiation	131
	1. General	131
	2. The Radiation Environment	132
	3. Internal Radiation Levels	146
	D. Vacuum	168
	1. Description	168
	2. Behavior of Material in High Vacuum	168
	3. Summary	171

CONTENTS (CONT)

Chapter		Page
6	MAN'S CAPABILITY TO PERFORM SPACE MAINTENANCE	
	TASKS	173
	A. General	173
	B. Task Requirements	173
	C. Motion and Manual Dexterity	185
	D. Extravehicular Suit Restrictions	192
	E. The Visual Environment	199
	1. Location of the Target Vehicle	199
	2. Visual Adjustments at the Work Area	202
	3. Effects of Weightlessness on the Visual Sense	216
	F. Weightlessness	220
	G. Discussion and Recommendations	220
	1. Workplace Provisions	221
	2. Provisions for Proper Lighting	221
	3. Provisions for the Weightless Environment	222
7	SPACE MAINTENANCE CONCEPTS	223
	A. General	223
	B. Automatic Maintenance Systems	229
	1. System Replacement	229
	2. Automatic Self-Maintenance	230
	C. Comparison of Automatic and Man-Centered Maintenance Systems	234
	D. Man-Centered In-Space Maintenance	237
	1. Special Purpose Maintenance Vehicle	237
	2. On-Board Maintenance Capability	238
	E. Conclusions	240
8	TOOLS AND FASTENERS	241
	A. General	241
	B. Task Requirements	241
	C. Special Tool Requirements	242
	D. Laboratory Exercises	244
	E. Conclusions	249
9	REMOTE MANIPULATORS	251
	A. General	251
	B. Task Requirements	253
	1. Maintenance Checkout, and Repair	253
	2. Space Station Assembly	255

CONTENTS (CONT)

Chapter		Page
	3. Materials Transfer	256
	4. Emergency Assistance	257
	5. Manipulator Operations	257
	C. Manipulator Control and Actuation	261
	D. Manipulator Feedback	265
	1. Visual Feedback	265
	2. Force Feedback	266
	E. Environmental Considerations	266
	1. Hard Vacuum Compensation (Cold Welding)	267
	2. Techniques of Operating Mechanisms in the Space Environment	268
	F. Manipulator Tools and Accessories	269
10	SUMMARY AND CONCLUSIONS	271
	A. Summary	271
	B. Conclusions	272
	1. Space Systems	272
	2. Subsystem Analyses	273
	3. Malfunction Diagnosis	274
	4. Maintenance Concept Development	275
	5. Task Analyses and Maintenance Design Handbooks	276
	6. Maintenance Worker Encapsulation	277
	7. Remote Manipulators	277
	8. Tools and Fasteners	278
11	REFERENCES	280

ILLUSTRATIONS

Figure		Page
1	Space Maintenance Techniques - Study Plan	2
2	General Configuration - Earth Satellite Weapons System	8
3	System Breakdown - Earth Satellite Weapons System	9
4	System Breakdown - Earth Satellite Weapons System	10
5	General Configuration - Manned Military Test Space Station . .	12
6	General Configuration - Space Station Supply Vehicle	13
7	Space Maintenance Capsule Configurations	16
8	Rigid Boom - Side Location	25
9	Typical Capsule Structure	35
10	Typical Structural Section	35
11	Logistic Vehicle - Structural Cross Section	37
12	Lunar Landing Vehicle - Structural Cross Section	37
13	Model 8101 Secondary Propulsion System	45
14	Agena Rocket Engine Model 8096	47
15	Model 8060 Reaction Control System (Automatic System Only) .	48
16	Model 8133 Advanced Agena Rocket Engine	49
17	LH ₂ -LO ₂ Rocket Engine	51
18	Typical Environmental Control System	66
19	Rendezvous Guidance Functions	80
20	Integrated Display System	81
21	Communications Functions	82
22	Attitude Control and Stabilization Functions	84
23	Meteoroid Flux Density	104
24	Micrometeoroid Impacts as a Function of Mission Time and Exposed Area	105
25	Micrometeoroid Impacts During Missions by Various Vehicles	106
26	Penetration and Spalling Flux versus Thickness of Aluminum .	110
27	Penetration and Spalling Flux versus Thickness of Aluminum .	111
28	Penetration Flux for a 4-hour Space Maintenance Capsule Mission	120
29	Electron Distribution in Outer Van Allen Belt	134
30	Estimated Intensity of Trapped Protons and Electrons	136
31	Isointensity Lines for Electrons	137
32	Isointensity Lines for Protons Ep7 > 40 MEV	138
33	Integrated Intensity of Electrons for a Lunar Mission (Outer Belt)	140
34	Integrated Intensity of Protons for a Lunar Mission (Inner Belt)	141
35	Integrated Intensity of Electrons for a 24-Hour Equatorial Orbit (Outer Belt)	142

ILLUSTRATIONS (CONT)

Figure		Page
36	Integrated Intensity of Protons for a 24-Hour Equatorial Orbit (Inner Belt)	143
37	Solar Flare Decay with Time	145
38	Gamma Dose From a Solar Flare	147
39	Integral Energy Spectra for Solar Flares, Inner Belt Protons, and Galactic Primary Protons	150
40	Proton Energy Spectrum Impacting 1 cm ² of Vehicle, (Lunar Trajectory)	151
41	Electron Energy Spectrum Impacting 1 cm ² of Vehicle, (Lunar Trajectory)	152
42	Dose Rate Behind Shielding for Solar Flare versus Time During Flare	154
43	Time Required to Execute Positioning Movement	186
44	Accuracy in Executing Position Movements	187
45	Speed of Cranking with Respect to Radii	188
46	Comparative Switch Positioning Reaction Time in Pressure Suit	195
47	Comparative Manual Dexterity Data with Pressurized Glove . .	197
48	Orbital Worker - Target Relationship	200
49	Typical Transfer with a Manually Operated Self-Maneuvering Unit	203
50	Range of Solar Illuminance	204
51	Dark Adaptation	207
52	Visual Acuity as a Function of Luminance	208
53	Visual Acuity with Relative Motion	210
54	Threshold Sensitivity as a Function of Preadaptive Luminance .	211
55	Successive Glare Effects	212
56	Flash Blindness Recovery Time	213
57	Simultaneous Glare Effects	214
58	Influence of Spares Interchangeability on Reliability Source . .	232
59	Reliability Comparison Manned - Automatic Systems for Rendezvous, Docking, and Assembly	235
60	Reliability Estimates	236
61	Air Bearing Floor Test Setup	246
62	Manipulator System	252
63	Memory Controlled Manipulator System (Man-Friday)	262

TABLES

Number		Page
I	Summary of Systems Breakdown	19
II	Terminal Rendezvous, Retrieval, and Docking Techniques	23
III	Propulsion System Requirements and Allocations	43
IV	Maintenance Requirements-Avionics Systems	86
V	Malfunction Sources	87
VI	Maintenance Tasks Selected for Detailed Analysis	88
VII	Maintenance Task Analysis	91
VIII	Space Station Assembly Task Analysis	94
IX	Maintenance Task Analysis (Propulsion)	96
X	Black Box Assembly - ESWS Tracking Transponder	97
XI	Component - Outer Wall of ESWS Manned Capsule	98
XII	Maintenance Task Analysis	99
XIII	Average Density and Modulus of Elasticity of Vehicle Wall Materials	114
XIV	Penetration and Spalling Particle Flux for Several Space Vehicle Structures	116
XV	Penetration and Spalling Particle Flux for Space Vehicle Structures with Bumper	119
XVI	Particle Masses Corresponding to Hole Diameters for a Given Rate of Pressure Drop	123
XVII	Estimated Time Intervals Between Critical Penetrations	124
XVIII	Effect of Free Volume on Transient Change of State Caused by Energy Transfer From a Meteoroid	128
XIX	Radiation Sources for Various Space Missions	148
XX	Doses for Various Solar Flares	156
XXI	Proton Dose Rates	157
XXII	Electron Dose Rates	158
XXIII	Internal Vehicle Dose Rate	160
XXIV	Estimated Daily Dose Rates Within a Space Suit	161
XXV	Threshold Dose for Various Materials	163
XXVI	Degradation of Extravehicular Space Suit by Radiation Deposition in Suit.	164
XXVII	Times Required to Receive Specific Biological Radiation Dose During Four Space Missions	166
XXVIII	560 Km Orbit During Solar Flares	167
XXIX	Important Quantities of Exosphere	169
XXX	Distribution of Maintenance Activities Internal and External to the Vehicle	174
XXXI	Principle Maintenance Tasks	175
XXXII	Behaviors Required for Structural Repair	179

TABLES (CONT)

Number		Page
XXXIII	Behaviors Required for Replacement of Gas Generator Injector Gasket	182
XXXIV	Arm Strength Data in Normal Environment for 5th Centile . . .	189
XXXV	Results of Tests Conducted on AN Standard Aircraft Bolts Utilizing the Air Bearing Platform	191
XXXVI	Relative Luminance Values	205
XXXVII	Recommended Illumination Levels	209
XXXVIII	Environmental Parameters and Their Effects on Extra- vehicular Encapsulation Systems	239
XXXIX	Results of Tests Conducted on AN Standard Aircraft Bolts - Static Conditions	247
XL	Results of Tests Conducted on AN Standard Aircraft Bolts - Utilizing the Air Bearing Platform	247

CHAPTER 1

INTRODUCTION

This report presents the results of a study of the requirements for, and characteristics of, space maintenance activities. The study involved a major assumption: the optimum approach for accomplishing an efficient orbital space force in the next decade is one which includes the capabilities offered by trained maintenance personnel to assemble, maintain, and service the vehicles constituting the space force. This assumption had a significant effect on the studies conducted during the contract as the technical effort was directed primarily toward man-oriented maintenance concepts, rather than machine-oriented concepts. Consideration of alternate maintenance concepts was included however, and Chapter 8 discusses a number of maintenance concepts and presents a justification of the assumption.

A. OBJECTIVES

The specific objectives of the study were as follows:

- (1) The identification of the systems, subsystems and major components making up the Earth Satellite Weapons System, Military Test Space Station, an orbital logistics vehicle, a lunar landing vehicle, a space maintenance capsule and an extra-vehicular space suit.
- (2) The establishment of component failure rates and malfunction modes and cues.
- (3) The identification of maintenance requirements appropriate to restoration of the systems to an operable state.
- (4) The analyses of the expected environment and the orbital worker's capability of engaging in maintenance tasks in these environments.
- (5) The analysis of the tools, manipulators and equipments necessary for these operations and the identification of potential problem areas relative to their use in a space environment.
- (6) The development of preliminary maintenance concepts appropriate to the space maintenance tasks.

B. APPROACH

The methodological approach followed during the contract is presented in Figure 1. This figure represents the major study area and indicates the data integration steps which were followed.

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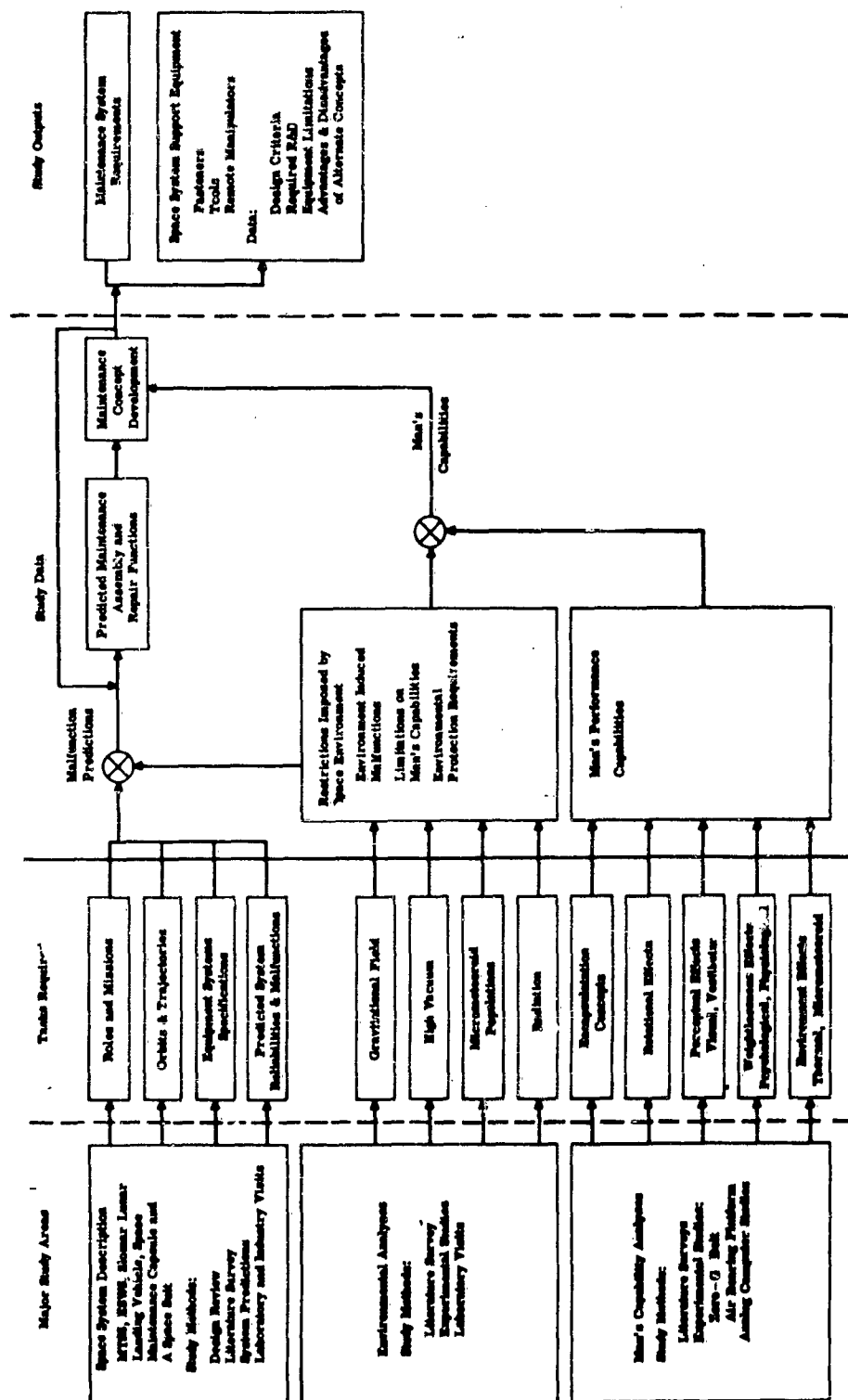


Figure 1. Space Maintenance Techniques - Study Plan

In view of the advanced state-of-the-art represented by those space vehicles requiring identification, it was evident from the outset that considerable effort would have to be expended in establishing conceptual models of each onboard system. Thus, an extensive literature survey was undertaken to collect and categorize, systematically, data appropriate to these space vehicles. Due to restrictions on the release of data defining in engineering detail the space vehicles to be analyzed, it was found that only limited information would be available. This situation restricted the scope of the identification phase to data which could be extracted from current periodicals and unclassified reports and papers such as those distributed by ASTIA.

These sources were studied, abstracted where potentially applicable and then filed under some forty subject headings appropriate to the space vehicles themselves, their subsystems, the encountered environment, human capabilities and activities necessary for activation and sustained operation of space vehicles. The principal vehical subsystems and/activation activities produced from the literature included:

- | | |
|--------------------------|---------------------|
| (1) Rendezvous & Docking | (5) Life Support |
| (2) Space Station | (6) Power Supplies |
| (3) Structure | (7) Avionic Systems |
| (4) Propulsion | |

An analysis of these data revealed that they can be separated into technical system categories which could then be integrated into usable and realistic models of each of the major subsystems making up the ESWS, MTSS, orbital logistics vehicle, space maintenance capsule and extravehicular space suit. To the extent possible, actual hardware systems were analyzed. The areas wherein such "hardware-based" maintenance analyses were possible included: propulsion, structures, and avionics.

Using these conceptual models, detailed component lists were prepared, quantities required in each system were established, failure rates estimated, and malfunction cues determined. Extensive use of schematics and hardware designs permitted accurate estimates to be made of malfunction modes, corrective measures to be undertaken, and the tool and fastener requirements. A typical maintenance requirement was selected from each subsystem analyzed and a detailed time line task analysis completed; thus establishing the time significance of the more prevalent space maintenance tasks and the areas wherein improvements in tools, fasteners, and design for maintainability could be recommended.

In many areas, the component malfunction modes were directly related to their operating environments. To aid in this identification, parallel studies were made to establish the environments within which the vehicles and their occupants would be operating. These studies were devoted to micrometeoroid effects; particle radiation effects on the space vehicles, their subsystems, and the human occupants; and vacuum effects.

Concurrent with the studies of the maintenance requirements and environmental effects, literature surveys were made and data collected on the human's capability to perform predicted maintenance tasks. Included in these investigations were weightlessness, mobility and encapsulation restrictions and visual/perceptual problems. As the data on the maintenance tasks became available from the maintenance task analyses, the human capabilities data was integrated and maintenance feasibility determined. The task of assigning realistic time intervals to the identified space maintenance operations required that knowledge of the decrement in dexterity caused by encapsulation and weightlessness be investigated. Some significant results were obtained by employing an airbearing floor for a series of maintenance task exercises. Typical tools, fasteners, connectors, and fittings identified in the maintenance analysis studies were selected and tests run to determine comparative time intervals for the fastening/unfastening task employing a variety of tools, with the worker in a single plane frictionless state. Tests were also run with the worker tethered to his work place, employing hand grip(s) and using a foot brace. Difference in time intervals and task difficulties were noted, with the worker employing socket type wrenches as compared with open end tools. Ambidexterity on the part of the maintenance worker was also found to be desirable.

Development of maintenance concepts required consideration of both the economic and the technical factors apparent in the manned space systems involved in this contract. The results from these preliminary analyses indicate that from a pure economic standpoint, a satellite replacement concept was most appropriate for the systems falling in the lower range of cost and weight. For satellites such as the ESWS and MTSS, onboard maintenance appeared most feasible. Thus, from a purely economic standpoint, a concept of manned maintenance of the space station appeared most attractive. This, then, became the basic ground rule for determination of concepts appropriate to the subsystems of the space vehicles. From the subsystem malfunction analysis it was determined that failures of an unscheduled nature tended to follow typical trends depending upon the functional categories involved. For example, the life support and propulsion systems whose components involved plumbing networks exhibited similar failure modes; e.g., leakage; structural components located in the space environment were mostly subject to damage from meteorite impact; avionic elements which malfunctioned required initially a troubleshooting sequence to isolate the faulty system, subsystem and finally component, then remove and replace activities and finally retest before reuse. In summary then, it developed that the maintenance concept for the larger satellites involved manned or onboard monitoring and correction of system malfunctions with replacement to a major component level considered most feasible.

C. STUDY LIMITATIONS

The study described in this report has definite limitations. First, only a few (5) space systems were studied primarily for in-space maintenance. While it was assumed at the outset of the study that these systems were representative of space

vehicles of the next decade. Further system studies, pointing out significant changes in some vehicle configurations, have since shown this assumption to be partially incorrect.

Second, only study-type data was available to the contractor for use in the study. Because of the generality of much of this data, it was necessary in almost all system analyses, to "engineer" the systems and subsystems with insufficient data. Because it was beyond the scope of the study to conduct detailed engineering studies of each technical area, it was necessary to make categorical decisions relative to subsystem design, operation, reliability, and accessibility.

At the initiation of the study, the assumption was made that man-oriented space maintenance would be the primary concern of the study; therefore, no extensive analyses of automatic maintenance systems were completed. Such studies are obviously necessary before a complete picture of the feasibility and economics of in-space maintenance and assembly operations can be drawn.

D. RECOMMENDATIONS

In order to enhance fulfillment of the developed maintenance concept and in consideration of the other analytical efforts associated with this contract, the following recommendations were made:

- (1) Expand the system identification and malfunction analysis to include such manned space systems as Gemini, Aerospace Plane, and Apollo, and unmanned boosters which may be employed for orbital rendezvous.
- (2) Based on (1) above, standardize malfunction correction concepts and measures.
- (3) As a parallel effort, establish a human capabilities index relating to the space environment.
- (4) Upon consideration of the data obtained in (1) through (3), initiate documentation of space vehicle design criteria and recommendations for use by design personnel in this field.

CHAPTER 2

SYSTEM DESCRIPTIONS

The principal objective of the Space Maintenance Techniques Contract was to establish, through systematic study, the maintenance, assembly, and repair functions of several specific manned space systems.

Selected for detailed study were:

Two multimanned space stations having orbital life expectancies of one to five years

An earth-to-orbit and return logistic vehicle

A manned lunar vehicle

A one-man space maintenance capsule

An extravehicular space suit.

It was evident from the outset that, in order to define the maintenance, assembly and repair requirements of these space vehicles, it would be necessary to determine the system mission profiles, establish data on general configurational approaches and finally, designate the subsystems which the vehicles contain. It was apparent shortly after initiation of the study that, due to the complexity of the space systems selected for study and their current development status, it would be necessary to conduct an intensive data accumulation survey during the first phase of the contract. This literature survey compiled a set of potentially applicable references and abstracts which were grouped into some forty categories containing over one thousand articles running from Apollo to Weightlessness. Sources examined during the survey covered some one hundred and twenty different unclassified periodicals from 1959 to current issues, all appropriate classified and unclassified reports and papers within Bell Aerosystems Company pertinent to space system description, and certain specific reports supplied by the contracting agency. The task of reducing this vast amount of information into a useable form was accomplished by employing a chart form of progressive system breakdown for each of the major vehicles of the contract. Once this breakdown was completed, appropriate approaches, system configurations, and component designs were extracted from the literature and functionally allocated to produce conceptual, but hopefully valid, models for analytical purposes.

The following data describe the systems studied during the contract and presents sufficient background information to permit a meaningful appraisal of the maintenance analysis sections of the report. (Chapters 3 and 4).

A. PERMANENT MULTIMANNED SPACE STATIONS

Two distinct categories present themselves when considering this type of space vehicle. First, would be the station designed to meet the requirement for an in-space platform where scientific research and military tests can be carried out; and second, the station whose missions would be primarily a strategic one, such as surveillance, reconnaissance, or bombardment. The systems most nearly meeting these mission requirements and which have received the most engineering study are The Earth Satellite Weapons System (ESWS) and The Military Test Space Station (MTSS).

1. Earth Satellite Weapons System (ESWS)

The ESWS is composed of manned orbital space stations at 300-mile altitudes. Logistic support and crew rotation for these stations is provided by a force of resupply vehicles, since a five-year orbital alert time is assumed.

Included in the ESWS is a space station and a logistic vehicle. Figure 2 presents the general configuration of this system. The ESWS space station element is comprised of three rather large sections, the six-man capsule which occupies approximately 12,000 ft³, and two launching platforms attached by booms to the capsule. The assembled station is approximately 185 feet long, 62 feet wide and weighs 38,000 pounds. Logistic vehicle docking facilities are provided on the station boom. The manned capsule is divided into two sections; one comprises the living quarters, and the second the working area. Two airlocks are provided at either end of the capsule for personnel ingress and egress. A large door is provided in the capsule working area section to permit admission of large vehicles for maintenance activities in a controlled environment.

To establish a convenient base line for predicting maintenance functions, the ESWS was divided into eleven major systems. Each of these has in turn been subdivided into types of systems having potential applications, and these finally have been examined to the point wherein actual components can be isolated. For ease of identification and to illustrate the breakdown employed for all vehicles, the ESWS family tree is presented in Figures 3 and 4. Similar system breakdowns were derived for all of the vehicles studied during this contract.

2. Military Test Space Station (MTSS)

In order to accomplish the MTSS missions, the station configuration must be selected to support:

- (1) research tests in most of the basic sciences**
- (2) celestial observations**
- (3) radiation belt charting**

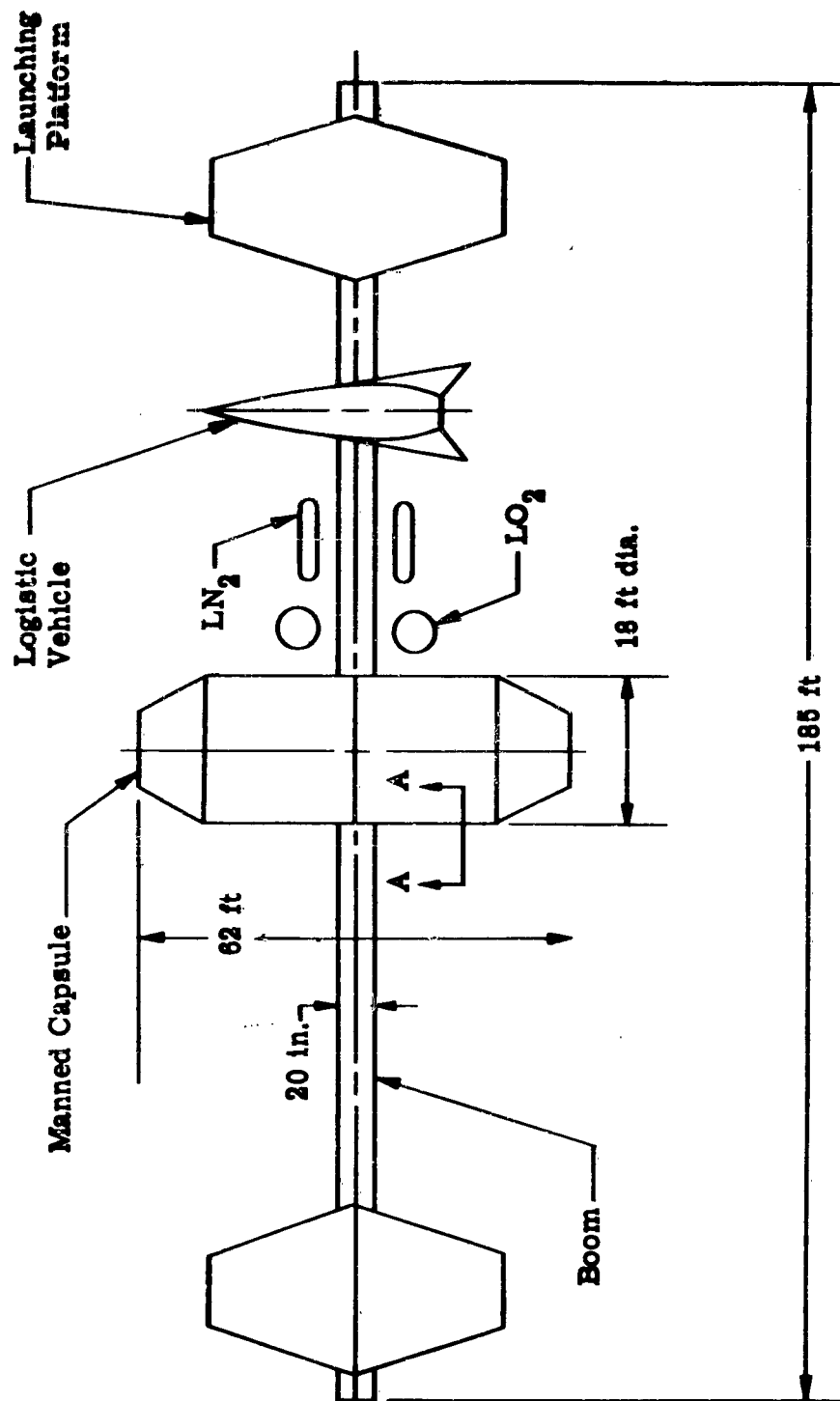
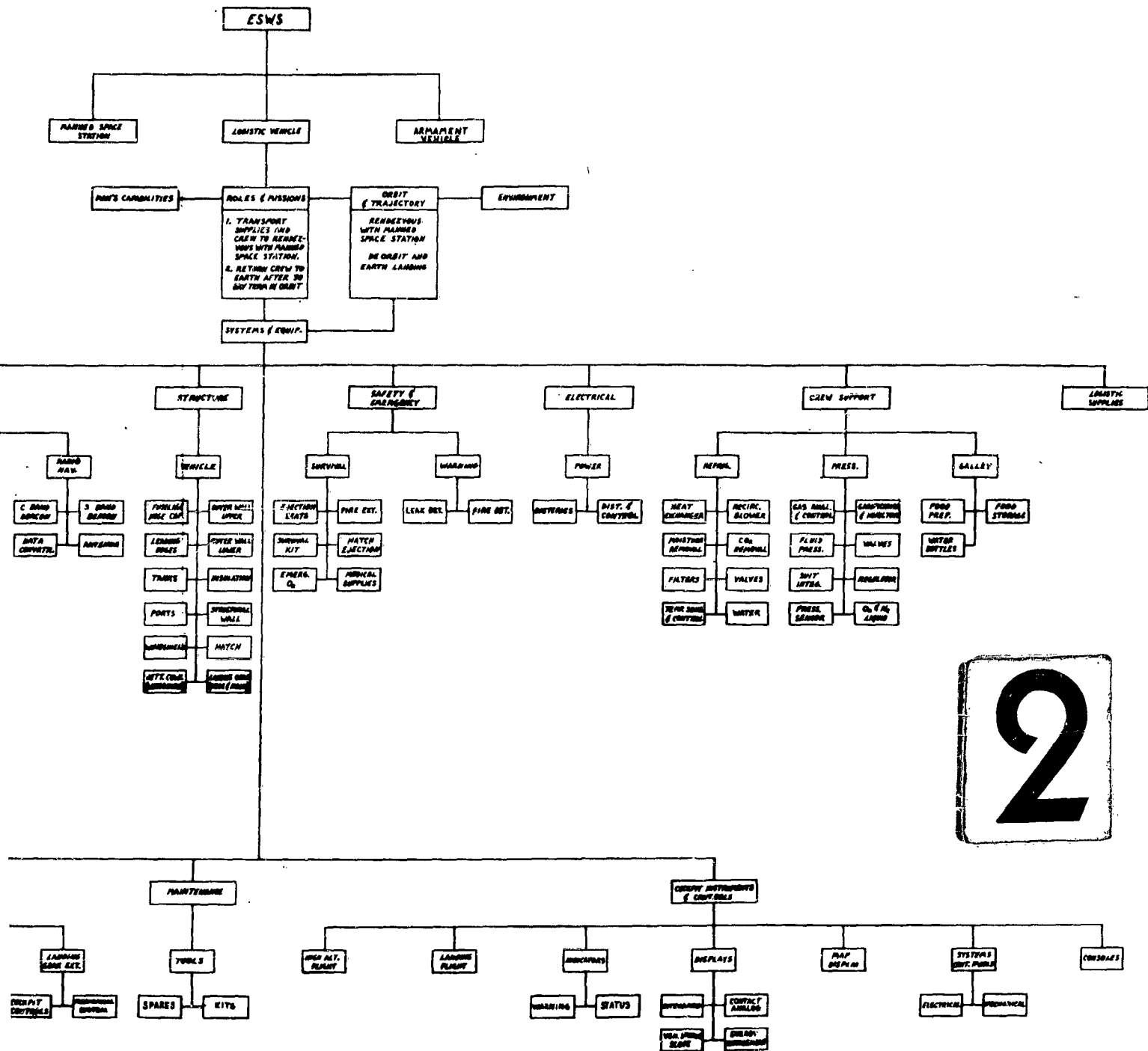


Figure 2. General Configuration - Earth Satellite Weapons System

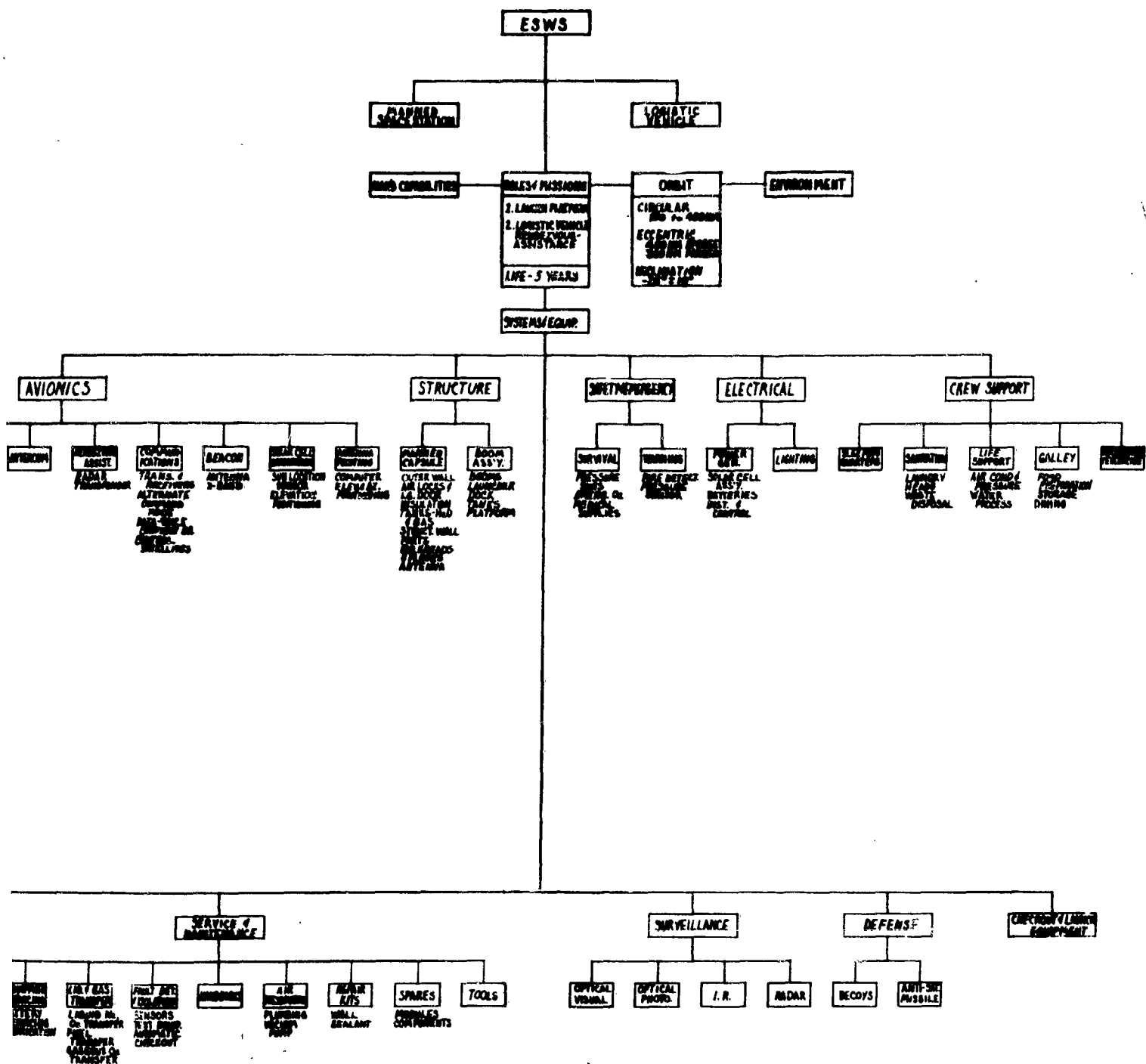


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Figure 3. System Breakdown - Earth Satellite Weapons System - Logistic Vehicle



10



Earth Satellite Weapons System - Manned Space Station



- (4) geophysical data collection
- (5) the testing of space components equipment and techniques of predominant military interest.

The MTSS is planned for the 1970 time period and will be placed at an orbital altitude of 300-500 miles. The functional lifetime will be a year or more with logistic resupply of supplies, equipment and crew on a scheduled basis.

The MTSS configuration, presented in Figure 5, is representative of this vehicle class. It is about 94 feet wide, 216 feet long, and weighs 441,000 pounds.⁽¹⁾ The station will be progressively assembled in orbit employing thirteen separate cylindrical sections, each 10 feet in diameter, 30 feet long and six 18-foot diameter spheres. These building blocks are joined to form a rectangular platform with cylinders lengthwise through the center. Additional sections containing the landing dock, entry hatch and medical laboratory, are attached to one center cylinder while the nuclear reactor and astronomical and physics laboratory are joined to the opposite cylinder. Rockets are provided at each extremity to spin the station about its longitudinal axis, thus simulating a 1 g gravity field in the outside compartments.

B. SPACE STATION SUPPLY VEHICLE

The operational feasibility of the permanent space stations described above is directly dependent upon development of a logistic network which will supply men and materiel to and from these complexes on a regular basis. The USAF as well as private industry have engaged in analytical programs for defining a vehicle which would satisfy the requirements. A considerable variance in configuration, concepts and approaches was found in the literature describing the various programs. However, certain design communalities were established due primarily to similar mission requirements in each case. For example, the majority of configurations examined incorporated wings for atmosphere maneuvering, heat sustaining features for the reentry phase, and gear for terrestrial landings. The crew compliment of these vehicles was found to vary from three, for the smaller simpler types, to six or more for the more exotic varieties.

The supply vehicle presented in Figure 6 is quite typical of this family of vehicles. This configuration features a highly swept, notched delta wing mounted to the top of a cylindrical fuselage. The outboard tips of the wing are drooped for directional control and a three-wheeled retractable tricycle landing gear is provided. The system equipment includes emergency and normal egress hatches, propulsion systems for orbital maneuvers, reentry and atmospheric cruise, life support apparatus, and extensive avionic gear for rendezvous and guidance control. The structural approach employs high temperature material such as molybdenum, and René 41 in a double-walled insulated arrangement. Reentry heating of leading edges and nose section would be handled by current state-of-the-art ablative or high temperature refractory materials, such as those employed on ballistic missiles and the Mercury heat shield.

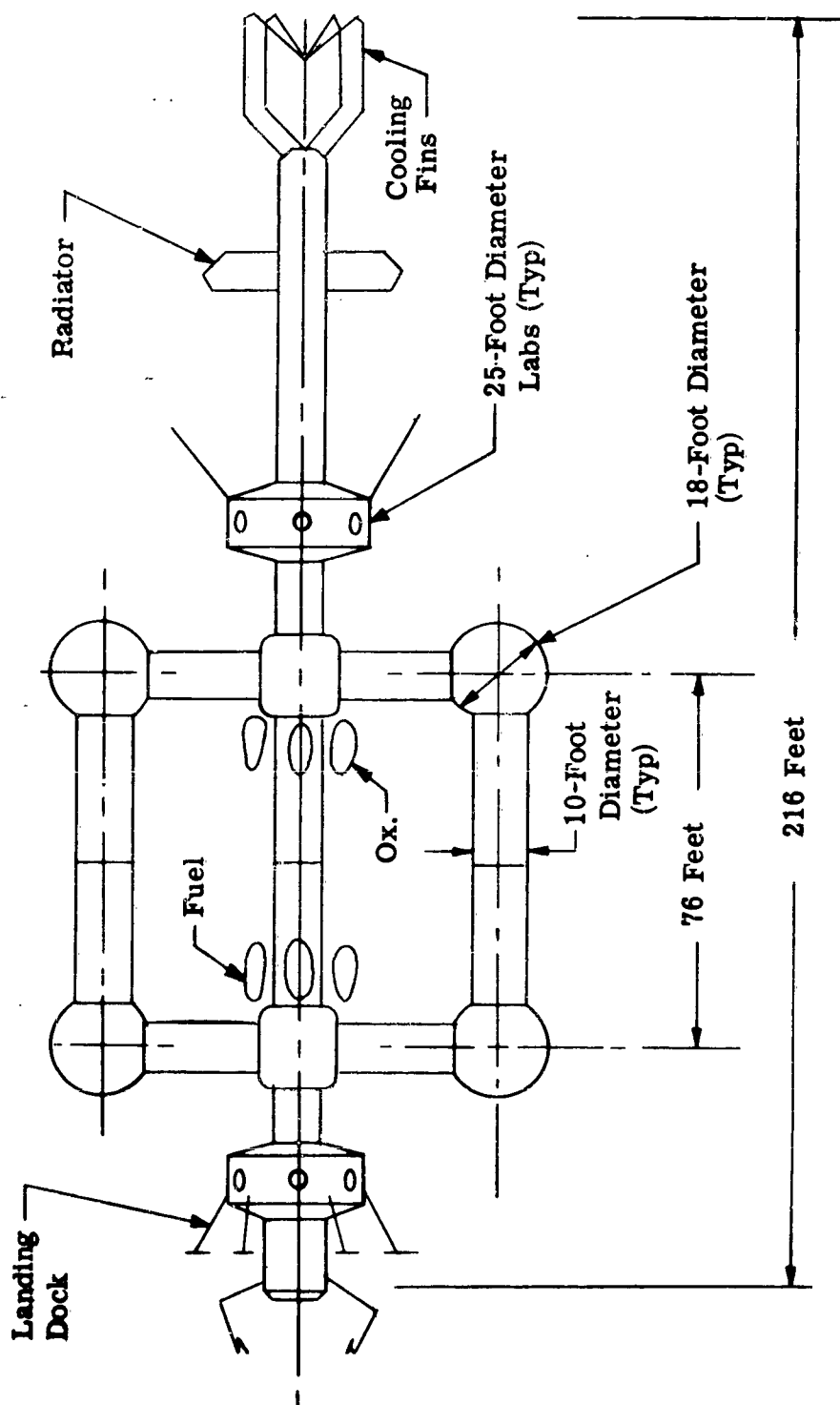


Figure 5. General Configuration - Manned Military Test Space Station,
Reference 1, Published with the Permission of the
Institute of Aerospace Sciences

* Adapted from Figure 1 of Kramer and Byers', "A Modular Concept for a Multi-Manned Space Station," Proceedings of the Manned Space Stations Symposium, Los Angeles, Calif. Apr. 1960

The overall length of this craft is 46 feet, with a maximum wing span of 28.3 feet and a gross launch weight of 21,200 pounds.

Another vehicle representative of this class is a landable and reusable vehicle intended for orbital operation on a scheduled basis. (2) This ferry can deliver a 14,000-pound payload in a rendezvous mission which may involve two to ten orbits over a period of three to twelve hours while supporting a crew of three. The configuration employs a 1000-square foot folding wing, a detachable manned capsule, reaction jets for maneuvering, inertia wheels for stabilization and control, and a 7500-pound throttleable rocket engine. The ferry is designed to reenter the earth's atmosphere using retro thrust and radiation cooled structure. A trusswork of thin gage nickel, cobalt or molybdenum is proposed with aerodynamic surfaces of thin gage corrugated sheet. The overall length is 78 feet 4 inches, unfolded wing span 44 feet 2 inches, with an orbit injection weight of 22,700 pounds.

C. LUNAR LANDING VEHICLE

This vehicle, of all of those considered in this contract, produced the least variety of concepts, primarily because study and development contracts had not been initiated when the system description studies were underway. Thus the configuration considered in the contract was composed of three basic Apollo elements: the Command, Service, and Lunar Excursion Module (LEM). The Command module carries the three-man crew together with mission control equipment, reentry provisions and life support equipment. The Service module contains the propulsion system for midcourse maneuvers, retro abort and escape from the lunar orbit as well as electric power and expendable supplies. The Excursion module will house two men during their descent to the lunar surface and return. It contains a complement of scientific instruments, avionics, and propulsion for maneuvering and lunar launch. The LEM module concept is ten feet in diameter, seventeen feet high on four legs, and weighs about 30,000 pounds. The Service module is 154 inches in diameter, 23 feet high, and weighs 46,000 pounds. The Command module is also 154 inches in diameter, 12 feet long, and weighs about 10,000 pounds. The launch configuration of the Lunar Vehicle will be about 60 feet long, and weigh between 85 and 90,000 pounds. A typical mission for this craft would extend for seven days, of which 2-1/2 days would be spent enroute and return, and 2 days would be spent for lunar exploration. The primary maintenance studies conducted on this vehicle during the space maintenance contract were devoted to analyses of the lunar landing module systems.

D. ORBITAL MAINTENANCE AND OPERATIONS SYSTEMS

Consideration of the orbital operations requirements resulted in the identification of four mission categories:

Consideration of the orbital operations requirements resulted in the identification of four mission categories:

- (1) Maintenance, Checkout, and Repair
- (2) Space Station Assembly
- (3) Material and Personnel Transfer
- (4) Emergency Assistance

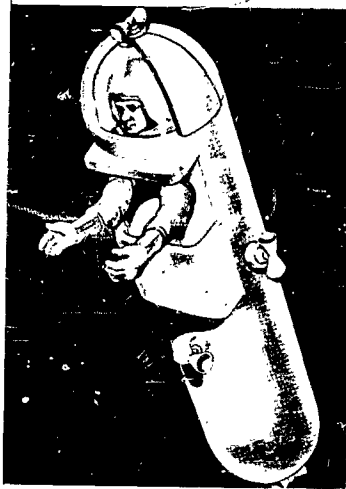
The accomplishment of these missions necessitates the requirement for the orbital maintenance worker to leave his mother vehicle and engage in tasks in the space environment. The extravehicular orbital worker system configurations can vary from a simple full pressure suit, with the required life support, propulsion, and stabilization and control systems attached to it, to the more complex rigid structure, encapsulating the man and including a well integrated combination of the required subsystems.

1. Space Maintenance Capsules

Numerous configurations for small capsules capable of accomplishing orbital operations tasks have been proposed. Figure 7 presents five of the most recent concepts. Each of the systems presented in this figure are one-man systems with the exception of the spherical vehicle concept proposed by Lockheed (Figure 7c). The vehicles range in weights from 175 pounds (Figure 7a) to 700 pounds (Figure 7c). In addition to these systems, others are described in Baker (6)

The Remora System proposed by the Bell Aerosystems Company is representative of this vehicle class, and in addition includes more available system data; as a consequence, this vehicle was the basic system incorporated into the contract. (7) The capsule is approximately six feet long, four feet in diameter and has an empty weight of 507 pounds. It has a four-hour mission capability with 100 percent reserve in life support and propellant supplies. A double-wall structural concept employing an inner magnesium shell, insulation core, and an aluminum alloy laminate outer wall is present. A meteorite bumper shield is suggested as a protective device for the worker and critical items of equipment. In order to accomplish the tasks allocated to this shuttle, provisions are incorporated for remote manipulators, pressurized gauntlets and grapples arms. The capsule life support system provides 100 percent oxygen at 5 psia with supplies sufficient for the four hour mission plus 100 percent backup. Safety features include provision for acceptance of a worker in a full pressure suit with its own life support system which would be activated upon failure of the capsule system. A thermal control system is also provided which maintains capsule temperature at 70°F. Other significant features include a reaction control and propulsion system, a control and stabilization system, a fuel cell-power supply, two-way communications, and status information display panel.

When transporting large logistic payload packages, space station modules, or booster assemblies, it appears desirable to attach more than one capsule to the



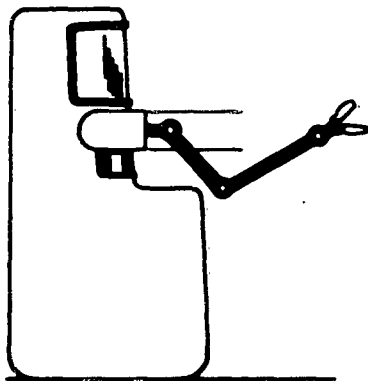
a. Cylindrical Concept with Fabric Armpieces, Lockheed Aircraft Corporation, Reference³.



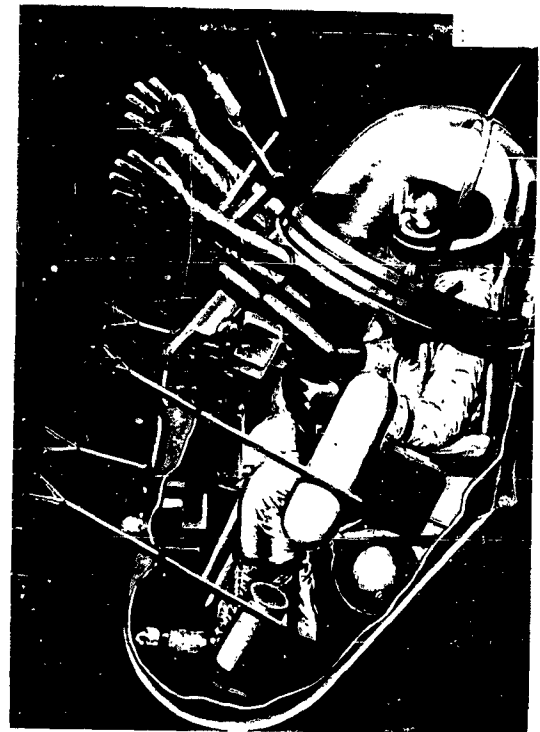
b. Cylindrical Concept with Manipulators, Lockheed Aircraft Corporation, Reference³.



c. Spherical Vehicle Concept with Manipulators, Lockheed Aircraft Corporation, Reference³.



d. Rigid Nonanthropometric Protection Schemes, General Electric Co., Reference⁴.



e. Remora, Nonanthropometric Encapsulation System, Bell Aerospace Systems Company, Reference⁵.

Figure 7. Space Maintenance Capsule Configurations

payload, thereby enabling faster transfer and rendezvous with the target. In situation the control, stabilization, and propulsion systems could be coupled capsule could control the required rendezvous maneuvers.

2. Extravehicular Space Suit

The space suit design selected for analysis consisted of a protective garment, an integrated life support system — including environmental control supply, waste control, and breathing supplies — locomotion provisions, stabilization and control, and communications (8). The protective suit consists of a multi-layered layup, including an undergarment-water pad composite, ventilation garment, bladder, restraint fabric, insulation layer and an outer reflective layer. Socks and gloves, and a partially soft helmet, complete the protective garment design. A removable pack containing the life support, propulsion, power supply, stabilization and communications systems is attached to the suit back. Pressure, temperature and radiation indicators are located on the suit-wrist area. Illumination is provided from removable chest and wrist lights. It should be emphasized that the extravehicular suit described in Belasco (8) represents a preliminary design based upon a requirement study. For example, while provisions for an orbital propulsion system were included in the design studies, they were hardly compatible with the present system designs presently completed. (5, 9, and 10) However, from the point of completeness of technical system study and design, this report represents the most extensive work conducted to date. The work currently in progress, under sponsorship, will result in the first pressure suit system designed for extra-vehicular use.

CHAPTER 3

SYSTEM IDENTIFICATION AND MAINTENANCE REQUIREMENTS

A. GENERAL

The success of prolonged manned space operations such as those proposed for the Earth Satellite Weapons System and the Military Test Space Station will be directly dependent upon man's capability to cope with failures occurring within the vehicle systems. Vast improvements can logically be expected in component reliabilities during the development periods which will precede these advanced space concepts. However, it is now generally accepted that anticipating, and then designing in, system maintainability characteristics can significantly add to the probability of mission success. It logically follows, then, that to provide the utmost benefit, a maintainability program should be formulated and activated during early system preliminary design phases. This contract is an illustration of the acceptance of this approach.

The methods employed for current maintainability programs have been standardized to a point where MIL type specifications have been prepared which establish the general requirements, program policies, and procedures. The methods established by these specifications have been well proven for terrestrial systems; thus it was decided to apply similar approaches oriented specifically to the space maintenance spectrum.

At the direction of the Air Force, analyses were made of the maintenance requirements of two major space systems (ESWS and MTSS), an earth-to-orbit and return space ferry vehicle (SLOMAR), an earth-to-moon and return system, and two space worker encapsulation concepts (space suit and hard capsule). Before estimates could be made of the maintenance requirements of these systems, it was necessary to obtain knowledge of the basic vehicle configurations and the subsystems and components involved. The previous section presented system descriptions of the six space vehicles studied during this contract. Table I shows a summary of the systems identified for each of the four major space vehicles studied. The seventeen systems noted in the table include a collection of minor subsystems which are identifiable with the principal category. The 41 subsystems noted for ESWS avionics, for example, include navigation, guidance, attitude stabilization, command and control, power supply orientation, antenna pointing and sensing, and computer facilities which would be associated with the space station proper as well as its logistic vehicle. It was obvious from the outset that detailed analysis of each of these subsystems would be impossible due not only to schedule considerations, but also to lack of related information in many of the areas identified. Thus, a decision had to be made regarding the procedure for conducting the identification task and the subsequent maintenance analyses. The rationale for establishing this procedure was as follows:

TABLE I
SUMMARY OF SYSTEMS BREAKDOWN

<u>Systems</u>	<u>ESWS</u>	<u>MTSS</u>	<u>Supply Vehicle</u>	<u>Lunar Landing Vehicle</u>
Avionics	41	9	14	13
Orientation	8	4	-	5
Structure	32	14	11	16
Hydro-Mechanical	19	2	5	3
Electrical	8	8	11	10
Environment Control	16	-	-	6
Propulsion	28	6	5	10
Safety and Emergency	13	6	2	5
Life Support	29	6	9	7
Instruments and Displays	14	7	8	9
Service and Maintenance	21	10	10	5
Surveillance and Defense	4/2	-	-	-
Checkout and Launch	3	3	-	6
Laboratories	-	21	-	-
Flight Control	8	5	5	12
Communication	10	6	4	5
Docking and Rendezvous	3	5	3	-

- (1) Each of the six space systems of this contract contain various forms of structure, avionics, propulsion, life support and power supply subsystems. These subsystems are not only common to each vehicle but also of prime importance for accomplishment of mission goals.
- (2) Studies of failure rate distributions accumulated during major missile programs indicate that the largest percentages of malfunctions occur in propulsion, avionics, and structures components in that order.
- (3) The vehicles of this contract will function, for the most part, in a true space environment where failures caused by vacuum, radiation, micro-meteorites, and temperature extremes can be expected. Avionic systems containing semi-conductors have high sublimation or decomposition rates in a vacuum. Radiation damage to these devices can also be expected. The flux density of meteorites will cause erosion, spalling, or puncture of space vehicle structure and exposed components. Evaporation of lubricants and cold welding of sliding surfaces may cause failures in mechanical components. High vacuum and temperature extremes will aggravate the leakage of fluid and gaseous plumbing systems.

On the basis of these significant facts, it was decided that a meaningful analysis could be completed by including the following principal space vehicle systems:

- (1) Propulsion
- (2) Avionic Systems
- (3) Structure
- (4) Power Supplies
- (5) Life Support Systems

Two additional technical areas were also deemed sufficiently significant for inclusion. Since the ESWS and MTSS may employ "assembly in space" techniques prior to operational activation, it will be necessary to provide equipment, material, and personnel for the initial rendezvous phase and for the docking and assembly activities. Thus, the techniques and components for accomplishing the following tasks were also included in the analyses: (1) terminal rendezvous, retrieval, and docking; and (2) orbital assembly, fastening, and sealing.

The next step in developing space maintenance requirements for the five major systems and the two orbital techniques noted above was to construct, from information in the literature pool, configurations which would fulfill the requirements of each category. Configuration or technique selection was based on:

- (1) Will the technique or component configuration selected meet the performance requirements of the vehicle to which it has been allocated?

- (2) Is the configuration within the state-of-the-art foreseeable in the time period of the space vehicle?
- (3) Have successful hardware or test programs been completed which demonstrate the feasibility of the configuration?
- (4) Is sufficient information available on the configuration to permit assessment of maintenance data including failure modes, malfunction cues, repair requirements and malfunction verifications?

By exposing each candidate configuration to the above criteria for acceptance, a single approach was selected for detailed analysis.

The representative configurations were constructed in sufficient detail to permit identification of the major component and/or component groups. In cases such as propulsion, life support, and avionics, schematics were developed which illustrated components, black boxes, plumbing or electrical networks, and information flow. Applicable geometric structural cross sections were constructed which identified materials employed, thicknesses required, and spacing of walls.

Completion of system identifications to this level next permitted collection of detailed component data including:

- (1) Quantities required
- (2) Mean component failure rates in terms of malfunctions per 10^6 hours.
- (3) Approximate operating times per mission.
- (4) Types of failures most likely to occur.
- (5) Cues to impending or existing malfunctions.
- (6) Procedure for verifying or isolating the malfunctions.
- (7) Maintenance requirements generated by each malfunction or failure.

The above data were collected without detailed consideration of the ways and means employable for accomplishing the required maintenance task. In view of the unique environment of these space vehicles and the potential information yield, it was deemed desirable to next undertake some detailed task analysis of selected maintenance requirements. A logical selection of tasks was made possible by applying a rank order of expected occurrence to the malfunctions and concentrating on those exhibiting predominance. The task analyses were then conducted on a time line basis and included detailed step-by-step accomplishment of the maintenance task, tools and fasteners encountered, force requirements, and recommendations for further investigations and design changes.

The following sections of this chapter present significant excerpts from the data generated in the manner described above, and Chapter 4 contains details of the maintenance task analyses conducted during the program.

B. TERMINAL RENDEZVOUS, RETRIEVAL AND DOCKING

The feasibility of complex multimanned space stations such as the Earth Satellite Weapons System and the Military Test Space Station is dependent upon development of earth-to-orbit and intra-orbit rendezvous techniques. These stations are too complex in terms of weight and volume to permit simultaneous boosting into orbit of all the sections. In addition, continued operation of these stations on a yearly basis will require crew rotations and resupply of material by scheduled earth-to-orbit ferry vehicle rendezvous. The normal or emergency maintenance of unmanned communications or inspector satellites will also generate the need for rendezvous functions. Because of this very basic functional requirement, it is deemed necessary to examine the techniques and equipments to be provided for accomplishing the rendezvous maneuver, and to develop the maintenance analysis generated by these devices.

The rendezvous maneuver, in a broad sense, covers all aspects of effecting a direct joining or meeting of two vehicles in the space environment. Three phases, established principally by separation distances, can be considered. The homing phase which extends from a few hundred feet to several miles, the docking phase which is analogous to the maneuvers of an ocean liner or the last few hundred feet of closure, and finally the coupling phase, which constitutes attachment of the two vehicles to each other. Most proposed rendezvous techniques employ automatic or semi-automatic radar or infrared guidance systems to accomplish the homing phase. However, due to the fine terminal adjustments and local decision requirements during the docking and coupling phase, it is desirable to incorporate human capabilities into the loop. In view of the complex equipment required for the homing phases and since this equipment may also be employed for navigation and guidance functions, a separate section devoted exclusively to maintenance analysis of avionic components is included later in this chapter. The following section presents a maintenance analysis of the equipment needed to effect the docking and coupling phase of rendezvous.

Table II shows the docking and coupling techniques which were analyzed during the contract. In view of the similarities of concept and maintenance requirements which exist within certain of these techniques, they have been categorized into five groups and will be presented in this manner for descriptive purposes.

1. Mechanical Coupling Techniques (12)

This technique consists initially of a search operation employing radar equipment, then a fine tracking and alignment sequence using IR techniques and finally mechanical extension of engaging booms, engagement and retraction to eventual module contact and assembly. The components employed for the radar and IR phases of this maneuver are discussed elsewhere; however, the booms and their mechanical components are unique and will be described. To differentiate between component locations

TABLE II
TERMINAL RENDEZVOUS, RETRIEVAL, AND DOCKING TECHNIQUES

Technique	Manual	Man	Automatic	Manned Vehicle	
		Assisted		with	Orbital
		Automatic		Unmanned	Station
				Stage	
Mechanical Coupling Techniques					
1. Rigid boom-side location			x	x	
2. Screw jack			x	x	
Ejectable Missile Technique					
1. Passive dart			x	x	
2. Inflatable cable		x	x	x	
3. Harpooning		x	x	x	
4. Scanning		x	x	x	
5. Cable rocket		x			x
6. Suspended-cable		x			x
7. Magnadok			x	x	
Projectable Arm Devices					
1. Telescopic		x	x	x	
2. Inflatable		x	x	x	
Direct Manned Participation Methods					
1. Capsule	x	x		x	x
2. Space suit	x			x	x
Optical Methods					
1. SATRAC			x	x	x
2. Piloted	x	x		x	x

and functions, the terms "target" and "chaser" will be employed for the two space station modules which are to rendezvous and mate. Two extendable, side-mounted, external tubular booms are employed — one on the target and one on the chaser module. (See Figure 8) The chaser boom engaging end is conical in shape with IR sensing receivers spaced around the periphery. The target boom engaging end is spherical with an IR transmitter in its center. Pulsed IR transmissions are picked up by the chaser receiver, alignment error data is generated and fed into the positioning mechanism to accomplish physical engagement of the target and chaser boom ends. Boom contact triggers an explosive thruster within the chaser boom cone which actuates three grabbing arms surrounding the target boom sphere. Due to the spherical configuration employed during this initial lock-on, it is presumed that some misalignment of chaser and target end interfaces will result. Thus, three indexing and alignment arms are provided in the chaser conical housing which, when properly actuated, will cause the modules to line up for final engagement. The chaser boom end opposite the cone is pivot mounted on a motor driven turntable which traverses on a V-block slide driven by a cable and motor system. Excessive boom impact forces are absorbed in a ratchet coupling which permits the slide to be accelerated and damped as required. The target boom does not have full motion capability, but can only be extended and retracted.

Other mechanical coupling techniques were investigated including screw jack retrieval, hydraulic or pneumatic retracting hooks, telescoping probes and retrievable guy lines. The screw jack principle is quite representative of these techniques. It consists of three peripheral located worm geared screw jacks powered by a vane type gas motor. The jacks terminate in hooks which engage a flange on the module interface when a closure distance of about two feet has been reached. Activation of these jacks will draw the modules together and trigger explosive pins for final sealing of the assembled joint. Accurate initial alignment of the two modules is required for this technique.

2. Ejectable Missile Techniques (12, 13 and 14)

Seven approaches were investigated in this category of retrieval schemes. They all involve essentially similar principles of ejecting a missile by springs, a rotating drum or explosive devices from one module towards various types of targets on the other module. The ejectable device, be it a dart, a harpoon or a missile, tows a cable which is retrieved after target engagement to bring the two modules into contact. Various target types are suggested which employ, for example, magnets which attract the ejected missile, a netting covered inflatable mylar balloon into which the dart is entangled or an IR source which guides a seeker missile to engagement. The most significant components of these devices, include the targets and missiles, the cable pay-out and the retrieval mechanisms, aiming devices, module interface locking pull-up and sealing components, and the associated power supplies and control systems.

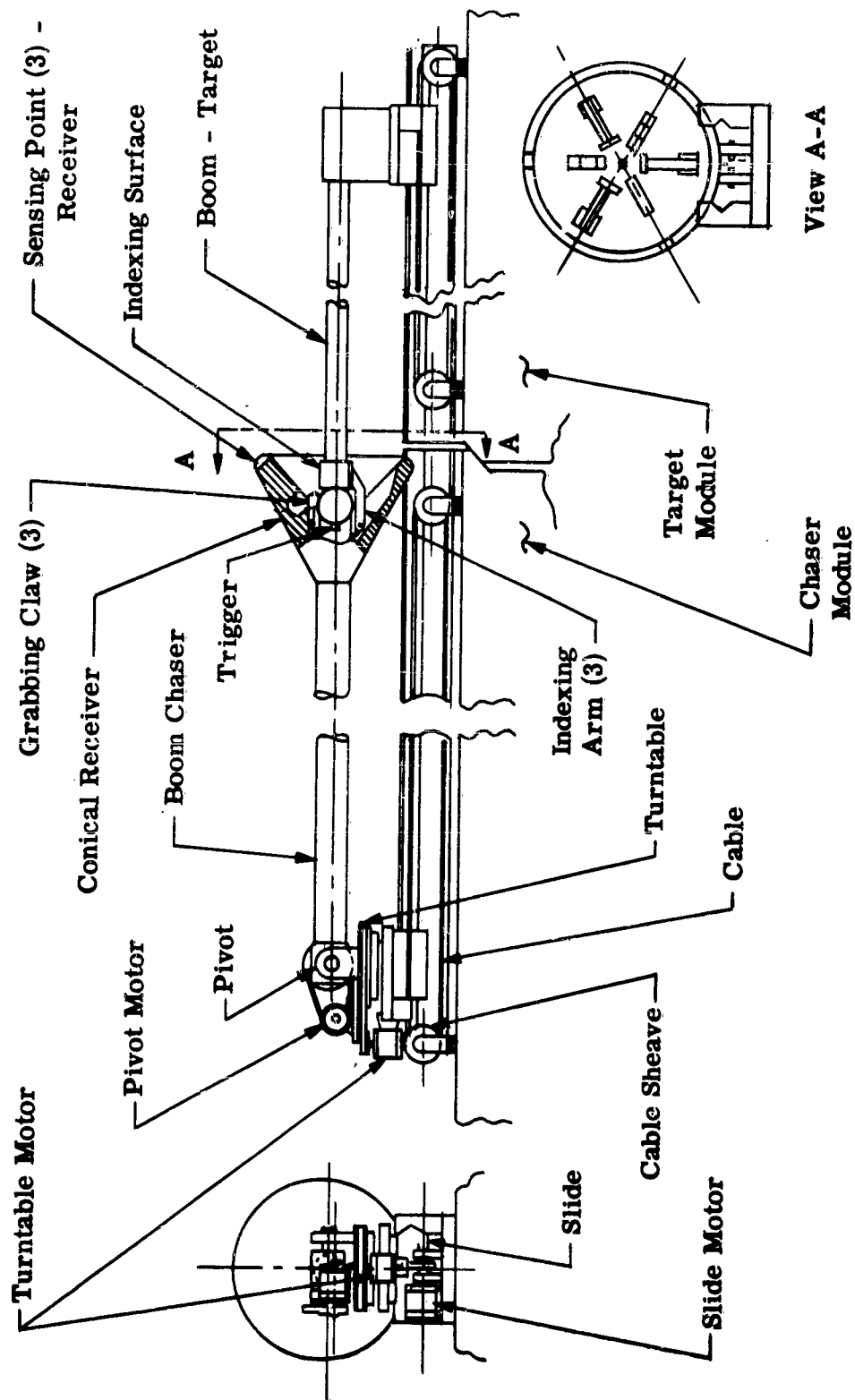


Figure 8. Rigid Boom - Side Location

3. Projectable Arm Devices (13)

Two different approaches were documented within this group of retrieval schemes. In each, an extendable boom or arm is projected from the chaser module toward a target on the target module. Extension techniques include a telescopic step-down inflated tube or an unfolded by inflation type tube. Targets include inflated net covered spheres and various active emanating devices. Retrieval of the modules is accomplished by reeling in the deflated arms and joining the interfaces.

4. Direct Manned Participation Methods (15)

The retrieval and docking schemes discussed thus far involve automatic or semi-automatic devices only. Two direct manned intervention techniques were also evaluated and documented during the contract. These were concerned with the retrieval and assembly of modules employing workers in space suits or capsules. In view of the complexity of the subsystems required of these vehicles, extensive detailed discussions are presented in the sections devoted to propulsion, avionics, structures, power supplies and life support. The manipulator requirements of the suit and capsule are discussed in Chapter 9.

5. Optical Rendezvous and Docking Methods (16, 17, and 18)

Two optical systems for rendezvous and docking were investigated. The first called SATRAC, for Satellite Automatic Terminal Rendezvous and Coupling, consists of optical homing guidance pods on booms which extend from the side of the chaser and target modules. The target module pod carries a light source and polarizer which emits a beam sensed by photo cells of a direction telescope in the chaser module. The chaser module also has a lamp, but without a polarizer, which is sensed by the photo cells of an error direction telescope. Orientation of the two modules is accomplished by polarizing the light radiated by the target vehicle and sensing the plane of polarization at the chaser vehicle. The pods are gimbal-mounted and servo pointed to the respective target lamps. The module axes are slaved by propulsion activation to the telescope axis thus setting and maintaining vehicle orientation during closure. A trapeze design is employed for direct engagement and coupling of modules.

The second optical method employs a ruby laser transmitter and a laser return signal receiver integrated with a photo multiplier, an amplifier and a gating circuit. By sensing the number of cycles completed during a timed interval of pulse transmission and displaying this information in digital form, range to contact information is supplied for pilot control of the rendezvous maneuver.

6. Maintenance Requirements

The scheme employing extendable booms on chaser and target modules is representative of many of the various techniques, systems and components currently proposed for in-orbit coupling of space station modules. Thus, a detailed analysis of the maintenance requirements of this system was conducted.

The methodology employed to establish the maintenance requirements consisted of first breaking down each system into principal components or subsystems, then establishing realistic failure data for these items and finally developing the functions necessary for malfunction correction and hence completion of vehicle mission. This task was made simpler and more meaningful when it was observed that the majority of component malfunctions could be attributed to one or more of the following ten discrete failure classes:

- (1) Loosening of component mount or material failure of mount and/or component. Surface distortion.
- (2) Shift in mechanical or optical alignment.
- (3) Physical jamming, rapid wear, erosion or brinelling of moving parts.
- (4) Environmental effects such as cold welding, development of high resistance joints, micrometeorite bombardment, change in solid state component parameters, outgassing, molecular breakdown, lubricant deterioration or change in insulation resistance.
- (5) Inadvertent operation of components caused by vibration, corrosion, oxidation or loosening of submodules, cards or components.
- (6) Shorts or opens within a component, burnout of filaments or arcing within or around a component.
- (7) Increase in contact or output noise.
- (8) Excessive drift of component characteristics.
- (9) Change in component electrical value submodule gain or signal bypass.
- (10) Saturation effects.

By classifying the failures which each identified component may experience, it now becomes feasible to establish the events which will occur as a result and the cues which can be employed for discriminating the malfunction. Once this has been accomplished, estimates can be made of applicable verification techniques and the maintenance tasks required for system restoration and completion of the mission.

Chaser Vehicle Components

a. Infrared Sensing Receivers *GFR - N.A.

(1) Failure Type

Boom position to extreme end of travel in azimuth and/or elevation as characterized by inoperative receiver; incorrect homing positioning.

* GFR - Generic failure rate per 10^6 hours of operation.
N.A. - Information is not available at this time.

(2) **Malfunction Cues**

Arm position indication display (cockpit instr. panel).

(3) **Malfunction Verification**

(1) Measure signal of suspected IR sensing receiver.

(2) Measure signal output of other two receivers and compare.

(3) Verify by checking servo unit parameters, and inputs to motors (assuming vehicle built-in capability).

(4) **Maintenance Requirement**

Replace IR sensing receiver assembly.

b. **Limit Switch Trigger GFR 0.24**

(1) **Failure Type**

Chaser boom contact established with target boom, separation occurred, grabbing arms and alignment arms not initiated, thrusters not fired.

(2) **Malfunction Cues**

Range tracking information.

(3) **Malfunction Verification**

Continuity check switch, extravehicular task.

(4) **Maintenance Requirements**

Replace switch

c. **Explosive Thrusters GFR - 0.21**

(1) **Failure Type**

(a) Chaser boom contact established with target boom, grabbing arms initiated, indexing and aligning arms not initiated.

(b) Chaser boom contact established with target boom, grabbing arms not initiated, indexing and aligning arms initiated.

(2) Malfunction Cues

Visual — periscope, television, etc.

(3) Malfunction Verification

Inspection of indexing and alignment arms position. Extra-vehicular task.

(4) Maintenance Requirements

Replacement of thruster not considered as an ideal solution. System design should include a high pressure air valve at the actuator to enable alternate means of operating indexing and alignment arms. (In all probability this capability is present for earth checkout of system operation in areas where thruster operation is not practical.)

It appears that holding of the target boom can be maintained by the indexing and aligning arms when the grabbing arms malfunction. Considering roll misalignment to be 10 degrees total, the indexing and alignment arms are oriented such that holding can be accomplished. To insure that separation will not occur in the case when orientation in roll is such that the indexing and aligning arms are in contact with the highest points of the indexing surface, the ball end of the target boom should be designed with a diameter greater than the indexing surface.

d. Servo Motors GFR 0.30

(1) Failure Type

Boom does not extend and/or remains stationary in azimuth or elevation.

(2) Malfunction Cues

Arm position indication display on cockpit instrument panel.

(3) Malfunction Verification

(a) Check IR sensing receivers operation, associated servo outputs, inputs to motors.

(b) Check extension/retraction signal output (assuming vehicle inbuilt checkout system capability).

(4) Maintenance Requirements

Replace motor(s) — check turntable bearings operation, check lead screw bearings.

Note: Turntable motors have belt pulley secured to drive shaft, extension/retraction motor has worm gear secured to drive shaft.

e. Position Switches GFR 0.5

(1) Failure Type

Boom position data to pilot display omits specific point information.

(2) Malfunction Cues

Arm position indicating display on cockpit instrument panel does not provide presentation of arm position.

(3) Malfunction Verification

(1) Check arm position indications at other points. (Error test signals induced by built in checkout system could exercise arm position to positive and negative extremes to check display functioning at other points.)

(4) Maintenance Requirements

No requirements for in-space activities.

f. Bearings GFR 0.50

(1) Failure Type

Slow or erratic arm positioning response; failure of arm to position in azimuth or elevation or to extend.

(2) Malfunction Cues

Arm position indication display on cockpit instrument panel.

(3) Malfunction Verification

Same as e(3).

(4) Maintenance Requirement

Replace bearing(s) — check motor operating condition.

g. Control Relays GFR 0.03

(1) Failure Type

Boom will not extend or retract; no response in azimuth or in elevation positioning.

(2) Malfunction Cues

Arm position indication display on cockpit instrument panel.

(3) Malfunction Verification

Same as e(3).

(4) Maintenance Requirement

Replace control relays.

h. Rate Damped Servo System GFR N.A.

(1) Failure Type

Azimuth, or elevation homing response slow, improper positioning or no response obtained.

(2) Malfunction Cues

Arm position indication display on cockpit instrument panel.

(3) Malfunction Verification

Same as e. (3).

(4) Maintenance Requirement

Replace malfunctioned module.

C. SPACE STATION ASSEMBLY

This section will present some of the techniques identified for assembly of multi-manned space stations. Since maintenance requirements may be generated by the types

of fastening devices and seals employed for this assembly, a section will also be devoted to these techniques.

1. Assembly-in-Orbit Techniques (1 and 19)

The conceptual configuration employed for the ESWS is composed of two dimensionally identical cylindrical modules, each forming, during launch, the transition structure between a Saturn S-4 booster and a logistic vehicle (see Figure 10). These modules differ internally since, after orbital assembly, one will form the crew living quarters and the second will become the working area. To minimize the amount of orbital assembly required prior to system activation, all module floors, ceilings, fixed equipment, ducting and wiring will be installed and checked out prior to launch. Orbital operations will thus be restricted to joining of the two modules, sealing of the interface and connection of lines and cables between sections. Operational activities of the station can commence after erection of the solar power supply, and assembly of the docking facility.

The MTSS configuration consists basically of an assembly of eleven cylinders and six spheres joined to form a station 216 feet long by 94 feet wide (see Figure 11). In view of the complexity of this station and the time and distances involved, it is proposed that two or more four-man space tugs be employed for construction activities. These tugs are complete space vehicles incorporating propulsion, communications, guidance, life support and manipulator systems. Assembly operations, employing these tugs, will consist of retrieving spheres and cylinders from their respective parking orbits, maneuvering to contact the appropriate interfaces and applying peripheral clamps over the end flanges. Joint sealing is accomplished by internally pressurizing the modules, thus forcing a seal into grooves of the interface. Permanent seals are later applied from inside the module.

2. Self-Erecting Space Stations (20, 21, and 22)

Two different techniques for self-erecting stations were investigated. The first configuration consisted of six cylindrical modules with alternate tops and bottoms hinged to each other and packaged in a bundle with longitudinal axes parallel. The top of this bundle is attached to a Mercury shaped capsule containing the crew members. The bottom of the bundle is attached to the booster by a separable joint. In-orbit erection is accomplished by releasing a restraining mechanism causing the cylinders to spring into a hexagonal shaped station. The capsule is then positioned at the hub of this six-sided wheel and three telescoping spokes added which connect it to the perimeter. Sealing of joints and connecting lines between sections is then accomplished and station operation begun.

A similar system employing the integrated approach of ground fabrication and in-orbit self-erection consists of a series of toroidal shaped segments telescoped to form a pie-like wedge. This wedge is attached to a booster with the escape capsule

mounted to its top. Automatic unfolding is accomplished in-orbit with the resulting vehicle being nearly circular in planform with the reentry capsule forming the center hub.

3. Convertible Space Stations (23)

A scheme for converting an expended booster into a space station has been investigated. This station employs an upper stage propellant tank which has been modified to contain an insulated cylindrical column through its center which houses the equipment necessary for orbital operations. The reentry vehicle with crew and equipment is mounted above the propellant tank. Station activation consists of venting and purging the tank of all remaining propellants and gases, then sealing the openings and pressurizing to the desired degree. Equipment mounted in the tank center column and in the reentry vehicle can then be removed and mounted in the propellant tank, solar cell power supply collectors erected and station operation begun.

4. Inflatable Space Stations (24)

In this configuration, the station components consisting of the folded and stowed inflatable sphere, the escape and reentry vehicle, and all necessary operating equipment are attached to the booster through an adapter structure. Once orbit has been achieved, the sphere cover is jettisoned, the sphere inflated and equipment mounted and activated.

D. ORBITAL ASSEMBLY FASTENING AND SEALING TECHNIQUES (12)

1. Pyrotechnic Devices

These devices include explosive pins, explosive driven spike joints, explosive bolts, pyrotechnic cord welding and pyrotechnic forming. Operation of these fasteners depends upon the firing of explosive charges after module mating, which expands metallic walls thus forming a secure joint.

2. Module Seals

Retaining the integrity of space station modules through reliable and positive joint seals is of prime importance to mission success. Seal selection must consider the environmental effects of vacuum, temperature ranges, and radiation. Current data indicate that polymeric compounds degrade in a vacuum by breakdown into volatile fragments making their usefulness as seals questionable. Some common elastomers such as silicone, styrene-butadiene, isoprene and natural rubber exhibit good vacuum behavior. (25) Some current unmanned space vehicles employ room temperature vulcanizing (RTV) silicone rubbers for sealing purposes. Applicability of these compounds to the space sealing task is not known at this time, however, some potential would seem to exist. Other techniques which have been proposed include 'O' ring

types, inflatable gasket elastomerics, crushable metallic, low temperature solder, foaming resins, pyrotechnique forming, welded, and magnetic.

E. SPACE VEHICLE STRUCTURES

The two multimanned space stations considered in this contract present requirements which encompass nearly all aspects of space vehicle structural design. The Earth Satellite Weapons System and the Military Test Space Station are made up of the basic orbiting vehicle itself, which houses permanently assigned personnel and equipment, a logistic or commuter vehicle for station resupply tasks, including the need to reenter and land, and personnel encapsulation devices such as space suits and assembly shuttles which permit accomplishment of extravehicular maintenance tasks. Each of these devices has unique structural requirements due to the roles, missions and environment which will be encountered. In view of the authenticity to be gained by applying hard design data to the conceptual vehicles, it was decided to allocate, wherever possible, tested and proven concepts to satisfy the diverse structural requirements of these space systems.

1. Structural Configurations

a. Earth Satellite Weapons System

The configuration shown in Reference 19 consists of a cylindrically shaped manned capsule 62 feet long by 18 feet in diameter to which are attached cryogenic storage tanks, radar and communication antennas, and docking/landing facilities for other vehicles. The capsule structure features the double-wall concept with temperature control coils bonded to the inside of the outer wall (See Figure 9). Double wall evacuated and insulated cryogenic tanks are mounted to the docking facility support boom. This boom is constructed of 20 inch diameter by 18 foot long sections of aluminum alloy tubes assembled in orbit. The launching/docking platforms consist of trapezoidal frameworks of aluminum alloy tubes and steel cables attached to the extremities of the main support booms. The completed station is approximately 185 feet long by 62 feet wide.

The structural approach employed for the ESWS logistic vehicle is discussed at a later time, since the requirements for this device have been taken to be similar in all cases considered.

b. Military Test Space Station

The MTSS configuration of Reference 1 is an assembly of 13 cylinders, each, 10 feet in diameter by 30 feet long, 6 spheres, 18 feet in diameter, two cylindrical laboratories 25 feet in diameter by 10 feet long, and a landing/launching deck. The cylinders and spheres are assembled in erector set fashion to form a rectangular shaped configuration with a long center compartment extending to either side and housing the laboratories, the launching/landing dock and the nuclear power supply. The composite MTSS is 216 feet long, 94 feet wide with a total exposed area of about 25,000 square feet.

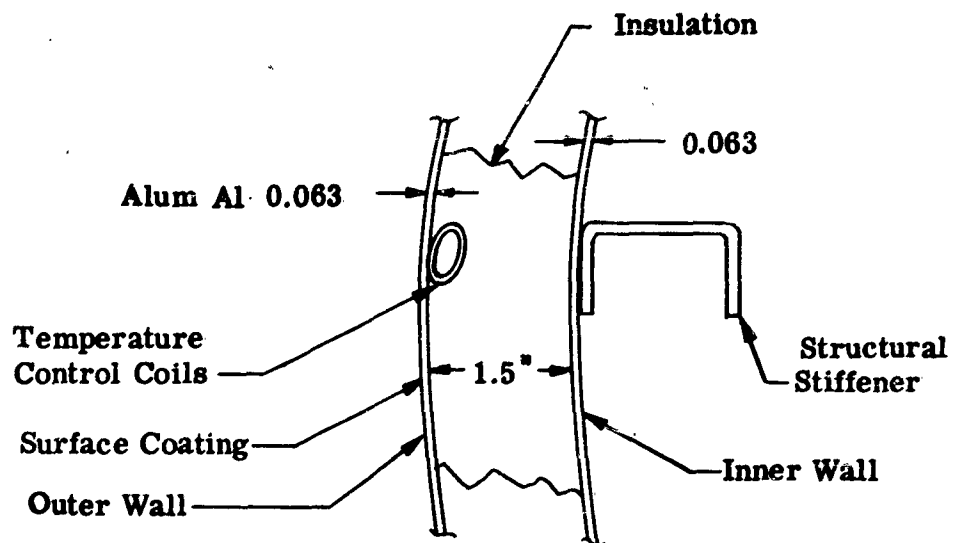


Figure 9. Typical Capsule Structure

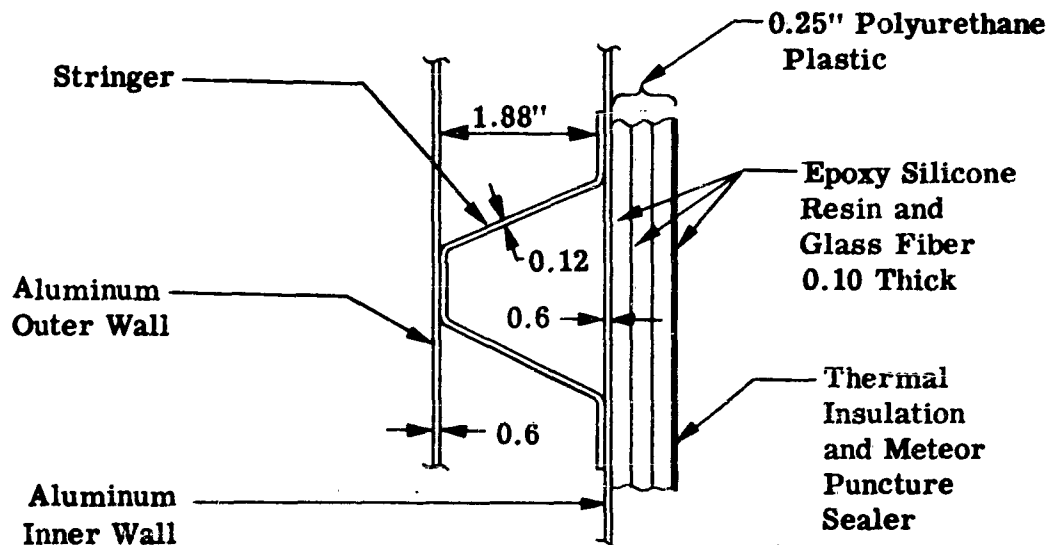


Figure 10. Typical Structural Section

The cylinders of MTSS feature double wall concepts with "Y" or hat shaped stringers separating the inner and outer walls. A thermal and meteor puncture sealer composed of three layers of polyurethane plastic and one layer of epoxy silicone resin and glass fibre are bonded to the inner wall (See Figure 10). Stringers and walls are composed of aluminum alloy sheets of various gages.

The MTSS spheres and laboratories are designed to support primarily internal pressure forces and because of their shape are most economically constructed of unstiffened double wall honeycomb material.

The nuclear power supply, though located remote to the manned portions of the station, is provided with a shadow shield of lithium hydride and lead. The cryogenic storage tanks containing propellants and life support oxygen are constructed of aluminum alloy in a double-wall insulated and evacuated configuration.

Some additional features of the MTSS which may generate maintenance requirements include glass viewing ports in the laboratories, TV cameras, IR illuminator, radar and communications antennas, airlocks, and searchlights..

Operational feasibility of MTSS as well as ESWS depends, initially, upon development of orbital assembly techniques and equipment and finally upon establishment of a logistic network for resupply of consumed materials and crew rotation. In the first case a manned assembly vehicle and probably an extravehicular space suit will be required and finally an earth-to-orbit and return ferry vehicle will also be in demand. These devices may well generate significant maintenance requirements thus a detailed discussion of their structural configuration is included in the following sections.

c. Logistic Vehicle

The structural configuration of this vehicle must incorporate provisions to permit earth-to-orbit rendezvous maneuvers, transfer from injection orbit to docking orbit and finally a launch from orbit, atmospheric reentry, and landing. The configurations investigated during this contract were, for the most part, winged, lift-flight aircraft. Their structures must be designed to support temperatures of 3500° to 5500°F in the region of nose caps and leading edges, and in addition, carry a pilot and crew with suitable thermal conditioning systems for their protection. A great variety of materials, concepts, and arrangements have been proposed as solutions for these structural problems. Most of these employ, for sections other than leading edges, a multiple layer concept consisting of heat sustaining outer panels, insulation blanket, and a cooled inner load carrying wall. Reference 26 compiles a number of different approaches which indicate the feasibility of employing current materials and techniques for the structure of the logistic vehicle. Figure 11 shows a cross section representative of the techniques which might be employed for a heat sustaining configuration.

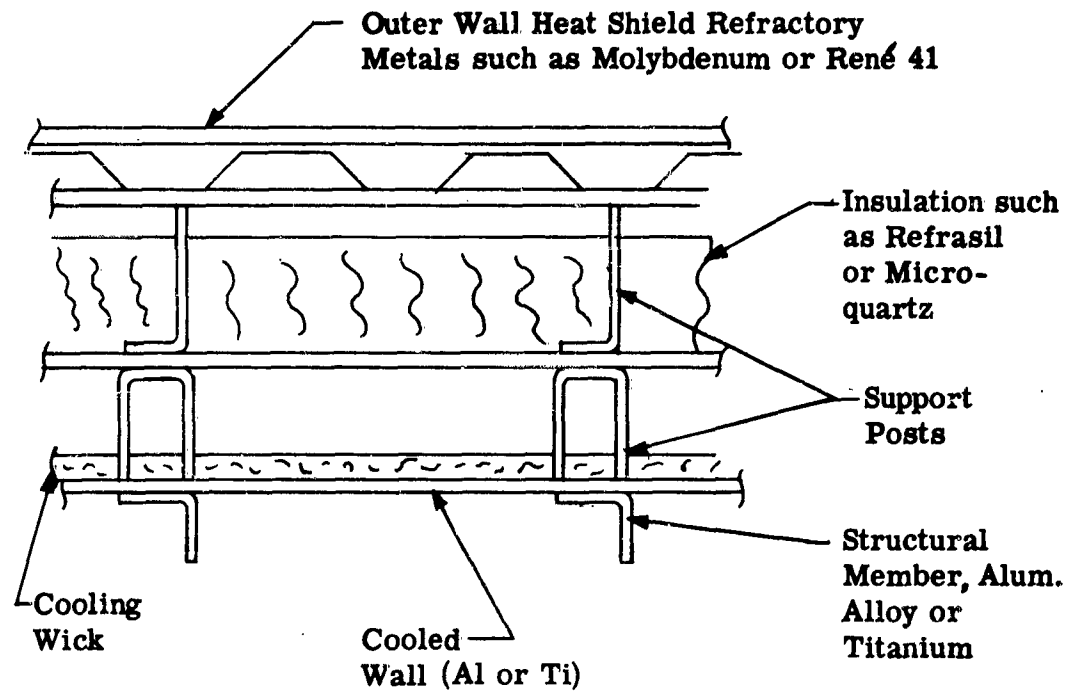


Figure 11. Logistic Vehicle - Structural Cross Section

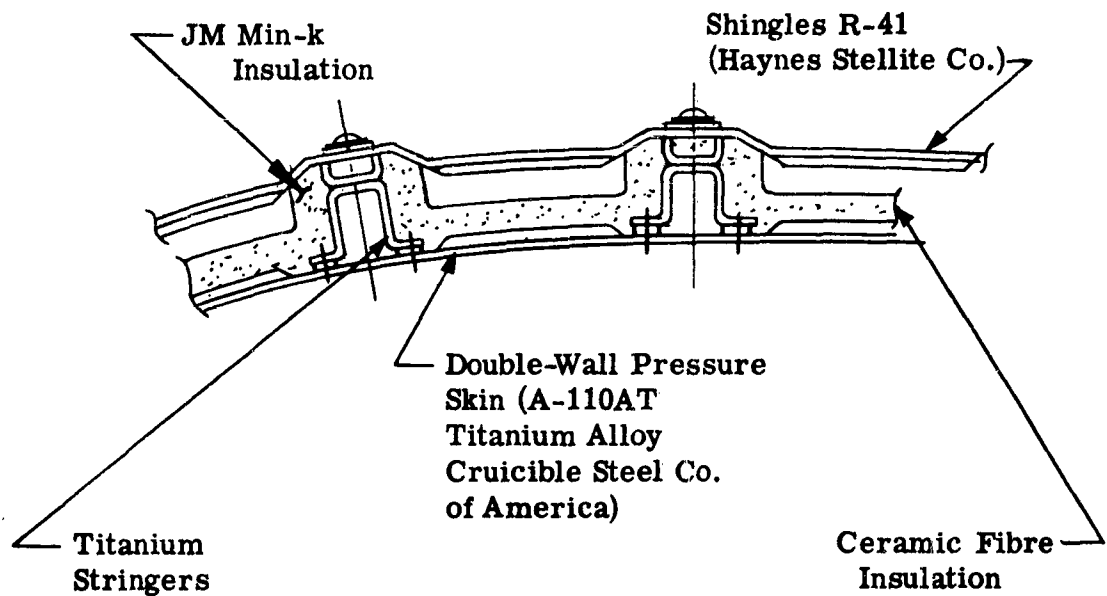


Figure 12. Lunar Landing Vehicle - Structural Cross Section

The nose cap and leading edges will encounter reentry temperatures from 3500° to 5500°F. Reference 26 discusses some configurations which have exhibited capabilities in this heat spectrum. Graphite and ceramic materials appear most useful with refractory metals such as molybdenum, columbium and nickel alloys like the René 41 series also competitive. Ablative materials such as phenolic-glass, silica or nylon have also been considered for application to the high temperature regions of manned reentry bodies. Since the exposed area of these regions is comparatively small, damage or repair requirements will not be included in this analysis.

d. Lunar Landing Vehicle

Each of the three modules making up this vehicle present somewhat different structural requirements. The Command Module which carries the three-man crew during the cislunar portion of the mission, must incorporate provisions for atmospheric reentry and recovery. The LEM, which is used for lunar landings and launches, will be employed only in a subgravity environment except for terrestrial launch accelerations. The Service Module, finally, is unmanned, does not land on the moon, nor the earth, and houses only propulsion, electric power generation and certain expendable supplies.

The structure proposed for the LEM consists of standard state-of-the-art lightweight high strength aircraft materials including various forms of titanium, magnesium and aluminum. The outer skin is likely to be of stainless steel or aluminum honeycomb with structural bulkheads and formers of titanium or magnesium. The lunar landing legs might be tubular aluminum with hydraulic or pneumatic shock absorption systems. A glass dome is anticipated for accurate maneuvering during lunar landing site selection and touchdown.

The Service Module structure will probably also feature current aircraft practice employing stainless steel or aluminum honeycomb with formed aluminum bulkheads and steel rocket engine supports.

The Command Module employs a Mercury type reentry and recovery concept with a similar configuration shape. The heat shield may, however, have to sustain longer and higher heating regimes for earth reentry, thus some new approaches must be incorporated. Early indications are that the shield will employ plastic reinforced charring ablation materials which have shown good characteristics during tests. The outer skin of the Mercury capsule double-wall is formed of a series of beaded shingles made from R-41, a nickel based alloy. The inner pressure wall is titanium alloy with titanium channel shaped stringers. A ceramic fiber insulation is employed between capsule walls. Figure 12 shows a typical cross section of this construction.

e. Space Shuttle

The one-man orbital worker capsule or space shuttle described in Chapter 2 features a double-wall structural concept made up of a magnesium inner wall, an insulation of Min-k 305 and an aluminum alloy laminate outer wall. Structural bulkheads and stringers are extruded aluminum box and formed sections. A network of tubing or tube sheet filled with coolant covers the outer skin for temperature control purposes. In view of the possibility of meteor encounter the space shuttle will be provided with a removable bumper shield surrounding critical areas. Consideration was also given to a self-sealing structural concept employing an elastomeric fluid between walls which would exude into a hole and seal by evaporation and subsequent hardening. A viewing canopy of glass is also provided.

f. Extravehicular Space Suit

The suit described in Chapter 2 is constructed of multiple layers of various materials. In sequence from the man outboard this layup consists of an undergarment, pressure bladder, restraint fabric, insulation layer and outer reflective layer. The materials investigated for potential application to the suit layup include cotton, nylon, bladder material, dacron link net, HT-1 padding and HT-1 Aluminized.

The boots proposed for this suit will be constructed of the same fabric layup described above, with soles and heels reinforced for improved wear characteristics. The gloves are removable and feature state-of-the-art pressure suit construction. The helmet is partially hard and includes an automatically actuated visor stack, antenna, and parts for feeding and vomitus removal.

2. Maintenance Analysis — Space Vehicle Structure

The structural concepts employed in the various elements of the Earth Satellite Weapons System utilize in one form or another, most of the geometrical arrangements, materials, cooling approaches and protective devices which were identified during this contract. Because of the diversity exhibited by this system it will be presented as an illustrative example of the techniques employed and results obtained during the structural maintenance analysis.

a. ESWS — Manned Capsule

(1) Failure Type

- (1) Puncture of outer wall or both walls by meteorites.**
- (2) Erosion of outer wall coating by micrometeorites.**
- (3) Damage to structure through impact during assembly or docking activities.**

- (4) Pressure damage such as leakages, blown seals and structure.
- (5) Crack propagation due to minute puncture, vibrational fatigue or creep.
- (6) Mechanical properties change due to long exposure to high vacuum.

(2) Maintenance Requirements

- (1) Repair puncture if sufficiently serious by applying a plug or patch to the hole.
- (2) Replacing the protective coating may not be feasible without complex factory equipment. Certain paints have been developed which might be applied with a space oriented applicator.
- (3) Minor bumping or denting of structure would probably be ignored. Repairs could be employed for more serious damage. Repairs to major structural bulkheads and frames would require removal of damaged section by cutting, forming a replacement from raw stock, securing the replacement and sealing the joints as required. This is a complex task not normally attempted.
- (4) Minor leakages would be corrected using a leak detector to isolate the area, then applying a sealing compound (preferably from inside) to the hole. Gross leaks would be plugged or patched. Replacement of blown seals is feasible if they are accessible and if internal pressures can be relieved during the replacement operation.
- (5) Discovery of cracks would prove difficult. Correction of serious cracks could be accomplished by applying a plug or patch. Stop drilling of cracks should be considered as a temporary measure.
- (6) The direction and magnitude of mechanical property changes is not well defined at this time. Designs may well have to consider this effect.

b. ESWS — Support Boom

(1) Failure Type

- (1) Meteorite puncture.
- (2) Assembly damage.
- (3) Damage due to docking of logistic vehicle.

- (4) Crack propagation due to minute puncture, vibrational fatigue or creep.
- (5) Mechanical properties change due to long exposure to high vacuum.

(2) Maintenance Requirement

- (1) Unless the puncture affects structural integrity, no maintenance action is deemed necessary.
- (2) Assembly damage would most likely occur at the boom ends where the joining clamps are to be applied. Minor damage in this area would be tolerated. Fracture of the boom wall by a docking logistic vehicle could be repaired in a manner similar to that employed for meteor puncture. Extremely severe boom damage might require complete replacement of one or more 18 foot sections.
- (3) Nicks, scratches and small dents might require filing out to prevent eventual flaw development and crack propagation.
- (4) Distortion of the boom may also require correction or complete boom replacement.

c. ESWS Launching Platforms

(1) Failure Type

- (1) Meteorite puncture.
- (2) Assembly damage.
- (3) Damage due to docking of logistic vehicle.
- (4) Crack propagation due to minute puncture, vibrational fatigue or creep.
- (5) Mechanical properties change due to long exposure to high vacuum.

(2) Maintenance Requirement

- (1) The diameters of the launching platform tubes are only 3 inches, thus it is considered uneconomical except in an emergency to repair any gross damage. A concept of complete replacement is recommended. In view of the small exposed area of the platform, the probability of meteor damage will be small. Breakage of the steel tube support cables might indicate the

need for a splicing task or again, complete replacement. Friction type cable splice fittings should be provided in addition to the usual tool kit.

d. ESWS Cryogenic Propellant Storage Tanks

(1) Failure Type

- (1) Meteorite puncture
- (2) Structural failure due to pressure buildup, damage during orbit injection or material properties change.
- (3) Crack propagation due to minute puncture, vibrational fatigue or creep.

(2) Maintenance Requirements

- (1) Puncture of the tank outer wall would probably be tolerated since the insulated area is evacuated to reduce cryogenic boiloff. Puncture of both walls would permit propellants to escape unless freezing and plugging occurred. Repair of a punctured cryogenic tank would be difficult. Inclusion of a puncture sealing compound might be the most reliable means of preventing losses.

F. SPACE VEHICLE PROPULSION

The first two major areas of system identification and the resulting maintenance requirements were devoted to space station rendezvous, retrieval and docking and space station assembly. The data presented in these sections were conceptual in nature since appropriate hardware has not been required by the current family of space vehicles. However, a somewhat different situation exists in other fields due to the potential application of current systems to future manned space vehicles. Propulsion systems, for example, have been developed which demonstrate sufficient reliability to suggest their use for certain tasks in the vehicles of this contract. In view of the value to be gained by employing the data generated in hardware programs, it was decided to select applicable existing rocket engines wherever possible for the maintenance analysis. Before an allocation of propulsion systems could be made, it was necessary to establish the space vehicle requirements. Knowing these requirements and the performance criteria of available engines, a systematic allocation was then possible. The vehicle requirements and the propulsion system selected to meet these requirements are presented in Table III.

TABLE III
PROPULSION SYSTEM REQUIREMENTS AND ALLOCATIONS

Earth Satellite Weapons System (19)

Logistic Vehicle

Orientation and range control B/A Model 8101

Rendezvous and deorbit B/A Model 8096

Space Station

Orbit adjust Saturn S-4

Orientation B/A Model 8101

Military Test Space Station (1)

Commuter Vehicle

Maneuver B/A Model 8101

Attitude control B/A Model 8060

Retro and Orbit transfer B/A Model 8133

Atmospheric cruise J-85 Turbojet

Space Station

Spin rockets B/A Model 8101

Orbit Adjust (Use assembly shuttle)

Stabilization B/A Model 8060 or
Ion engine

Assembly Vehicle

Thrust and Attitude B/A Model 8101

Logistic Vehicle

Retro and orbit transfer B/A Model 8133

Attitude control, micro-rendezvous and
maneuver B/A Model 8101

TABLE III (CONT)

Lunar Landing Vehicle

The propulsion requirements for the various elements of this vehicle could not be allocated to any known rocket engines, thus the data found in unclassified periodicals was employed to identify schematic arrangements and components.

Space Shuttle ⁽⁷⁾

Thrust and attitude combination

B/A Model 8101 or
Model 8060

Space Suit ⁽⁸⁾

Thrust and attitude combination

Cold gas ⁽⁸⁾
 H_2O_2 ^(9,10)

As indicated in Table III, the major propulsion requirements of the six space vehicles of this contract were satisfied by employing four existing Bell Aerosystems Company engines (Models 8101, 8096, 8060 and 8133), the Saturn S-4 and a J-85 turbojet. The Lunar Landing Vehicle requires the use of conceptual engines. An ion engine was also identified in connection with the orbit adjust requirement of the MTSS space station module. Detailed descriptions of these engines follow.

1. Propulsion System Descriptions

a. Model 8101

The Bell Aerosystems Company Model 8101 Secondary Propulsion System is a storable liquid bipropellant system designed to supply thrust on demand for a one year service period while in orbit. This system is currently employed in the Agena satellite program for assurance of a positive propellant head to the prime propulsion system and as a means of adjusting the orbit of the complete vehicle.

The system is composed of four thrust chambers, two 200 pound and two 16 pound, and a pressurized propellant storage and feed system. These engines employ hypergolic propellants consisting of mixtures of nitrogen and unsymmetrical dimethylhydrazine. A spherical source tank provides high pressure nitrogen gas to the pressurization system for propellant expulsion and valve actuation. Cylindrical tanks containing bladder expulsion systems provide storage for propellants and produce positive expulsion under space environment. Figure 13 shows a schematic of the system as it is currently employed.

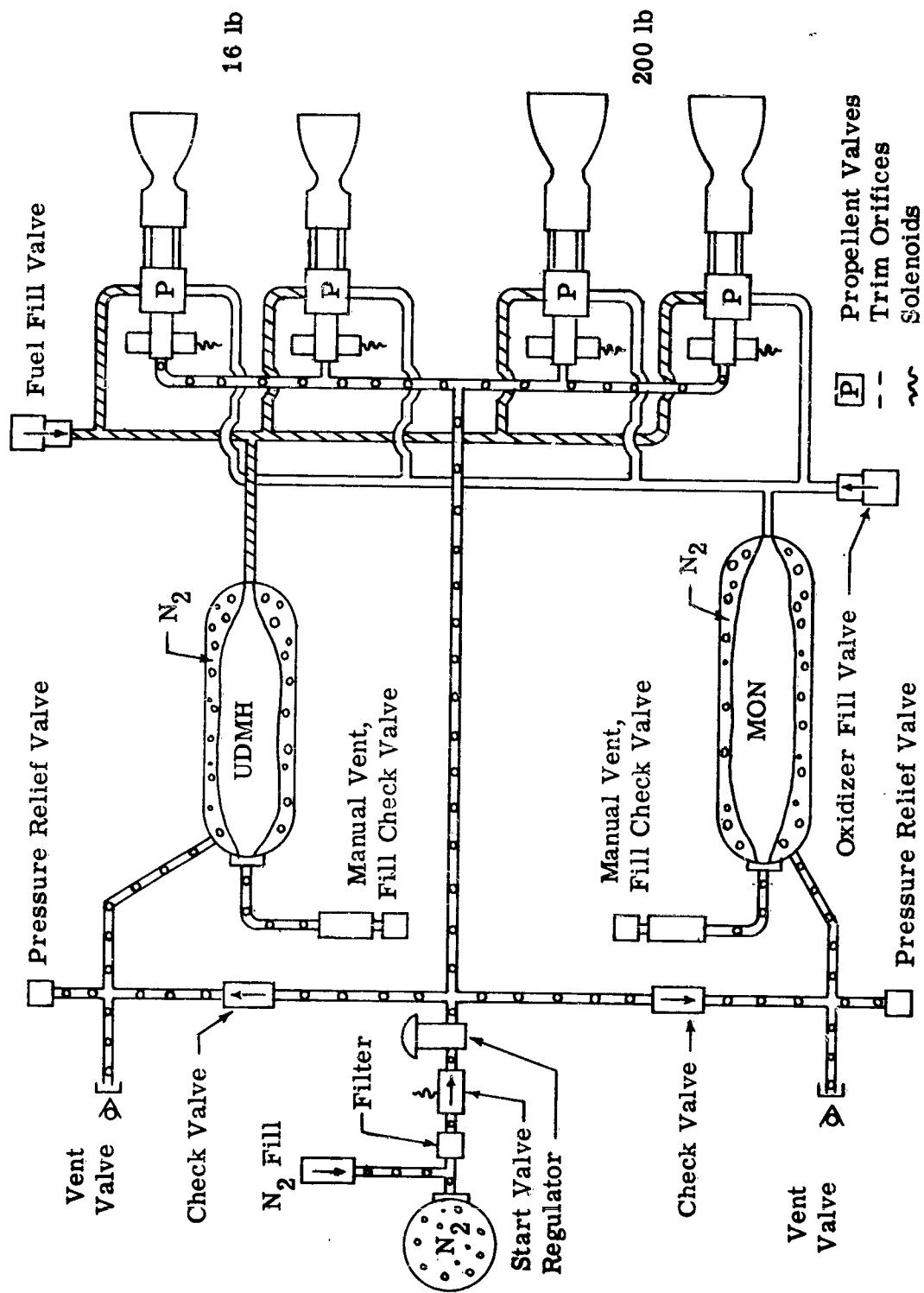


Figure 13. Model 8101 Secondary Propulsion System

b. Model 8096

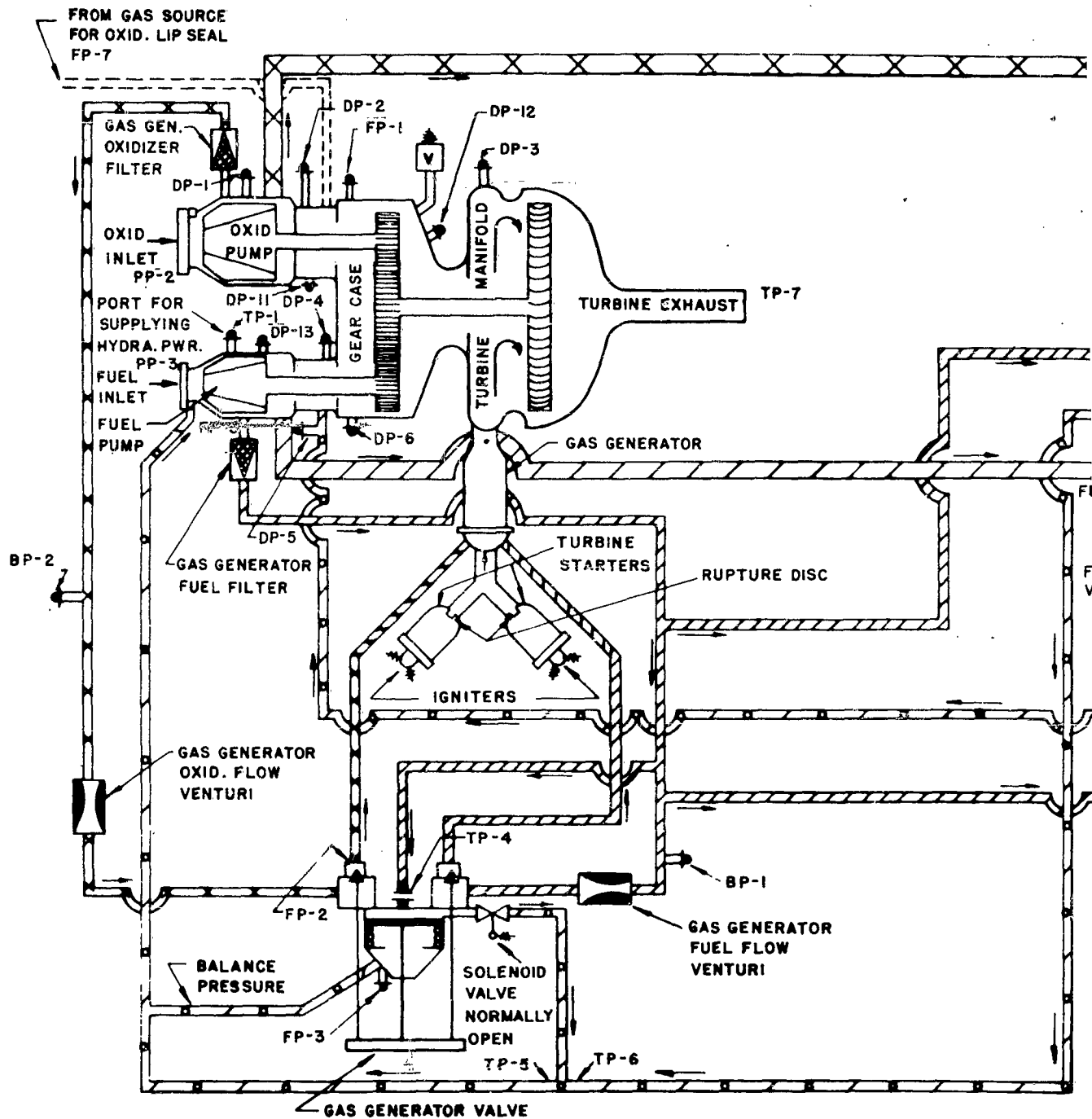
The Bell Aerosystems Company Model 8096 Rocket Engine is designed as the main propulsion unit for the Agena satellite. The Model 8096 engine produces 16,000 pounds of thrust with run durations of any desired combination up to a total of 240 seconds. In addition, a fast shutdown restart capability is provided. Inhibited red fuming nitric acid and unsymmetrical dimethylhydrazine are employed as propellants. The thrust chamber is regeneratively cooled with oxidizer flowing through passages in the walls. It is gimbal mounted providing coplanar motion and operates at a chamber pressure of 506 psia in a vacuum. A turbine pump is employed to deliver propellants from the tanks to the injector as well as to provide a power takeoff for operation of a hydraulic pump. Figure 14 shows a schematic diagram of this engine.

c. Model 8060

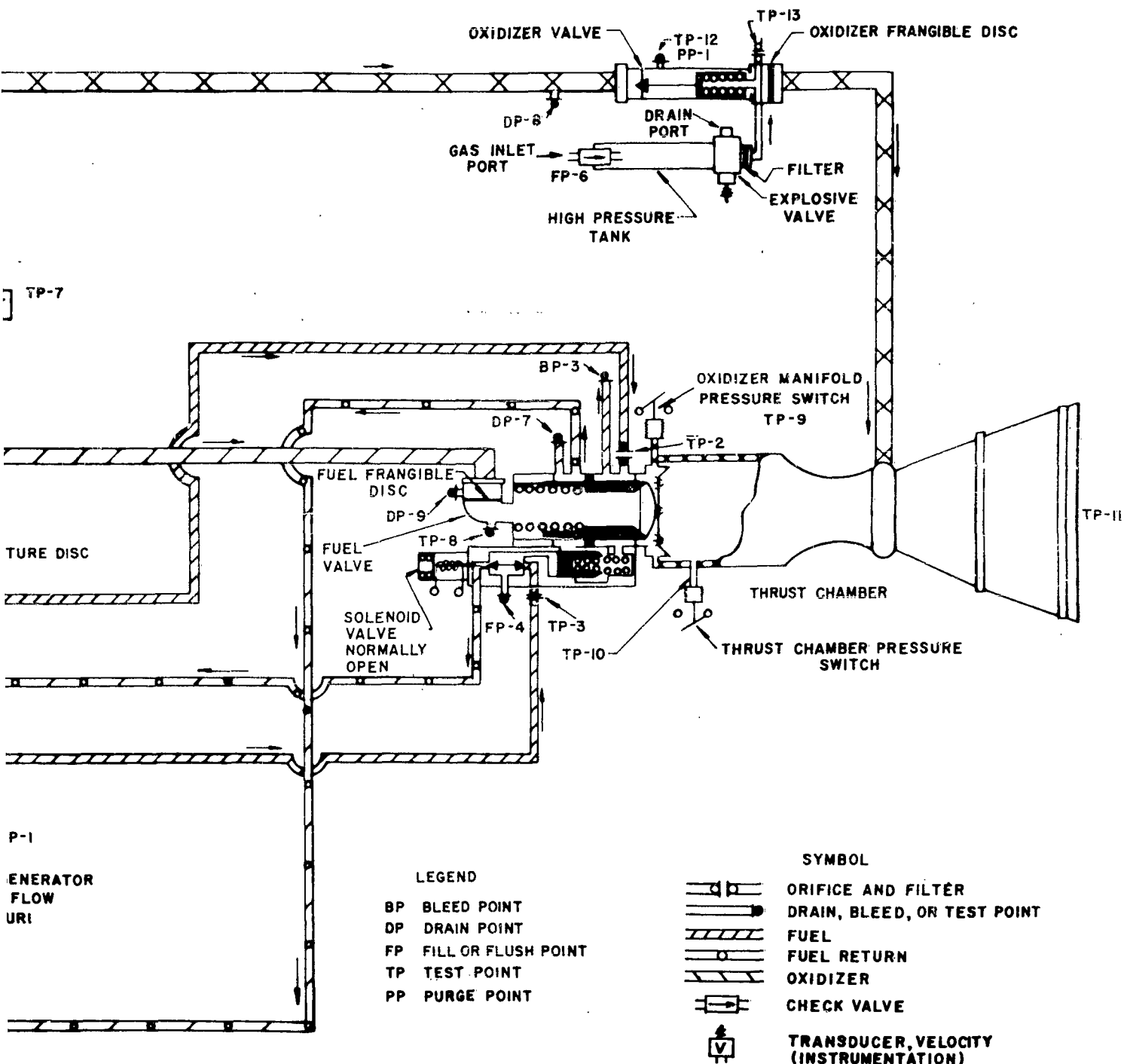
The Model 8060 Rocket System is produced by Bell Aerosystems Company for use as the reaction control elements on the Mercury capsule. Metered quantities of 90 percent hydrogen peroxide are decomposed in silver-plated catalyst beds in each of eighteen thrust chambers. The thrust levels provided in the Mercury system are 24 pounds, 6 pounds, 1 pound, and throttleable ranges from 0-24 pounds and 0-6 pounds. The pressurization system consists of a spherical tank filled with nitrogen gas at 2250 psi. This gas is directed through a pressure regulator and suitable plumbing to the bladder expulsion system of the peroxide tank. In order to be assured of mission success, two completely independent fuel supplies, plumbing and thruster systems are provided. Thus, even multiple malfunctions could be tolerated without catastrophic consequences. An illustrative schematic diagram of one of the Model 8060 systems is shown in Figure 15.

d. Model 8133

The Bell Aerosystems Company Model 8133 rocket engine is designed to provide improved restart capabilities and greater payload performance for Agena type missions. This system uses a 50/50 blend of unsymmetrical dimethylhydrazine and nitrogen tetroxide as propellants producing 16,890 pounds of thrust for durations up to 340 seconds per cycle. Four start stop sequences are provided. A turbine pump feed system is employed to transfer propellants from the tanks to thrust chamber injectors. Initial starting of the turbine pump is accomplished by first pressurizing two small start tanks with suitably regulated helium gas, then initiating the start sequence which ignites propellants and produces turbine speed build-up by gas impingement. Figure 16 shows a schematic of the Model 8133 engine.



1



2

Figure 14. Agena Rocket Engine, Model 8096

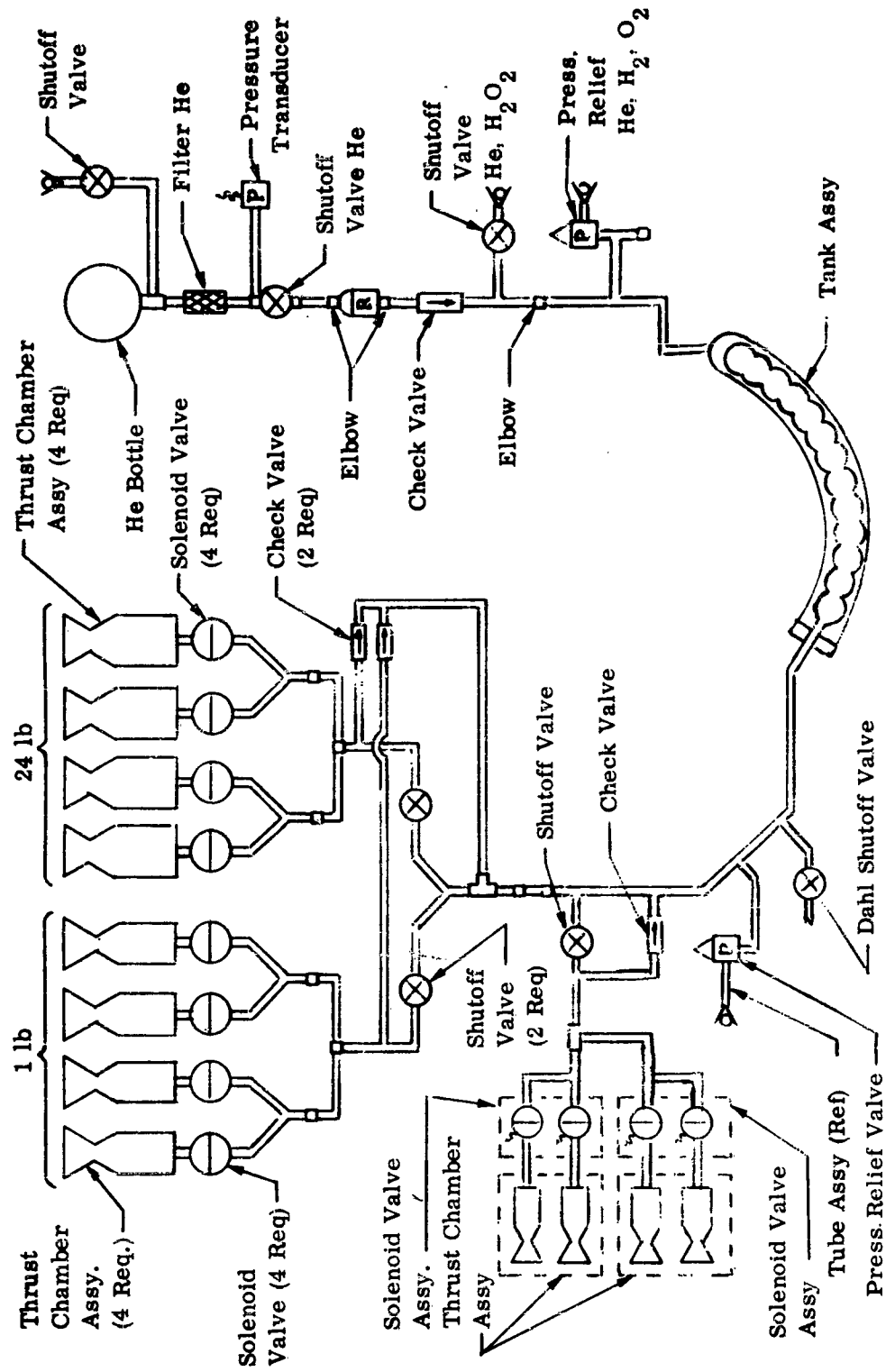
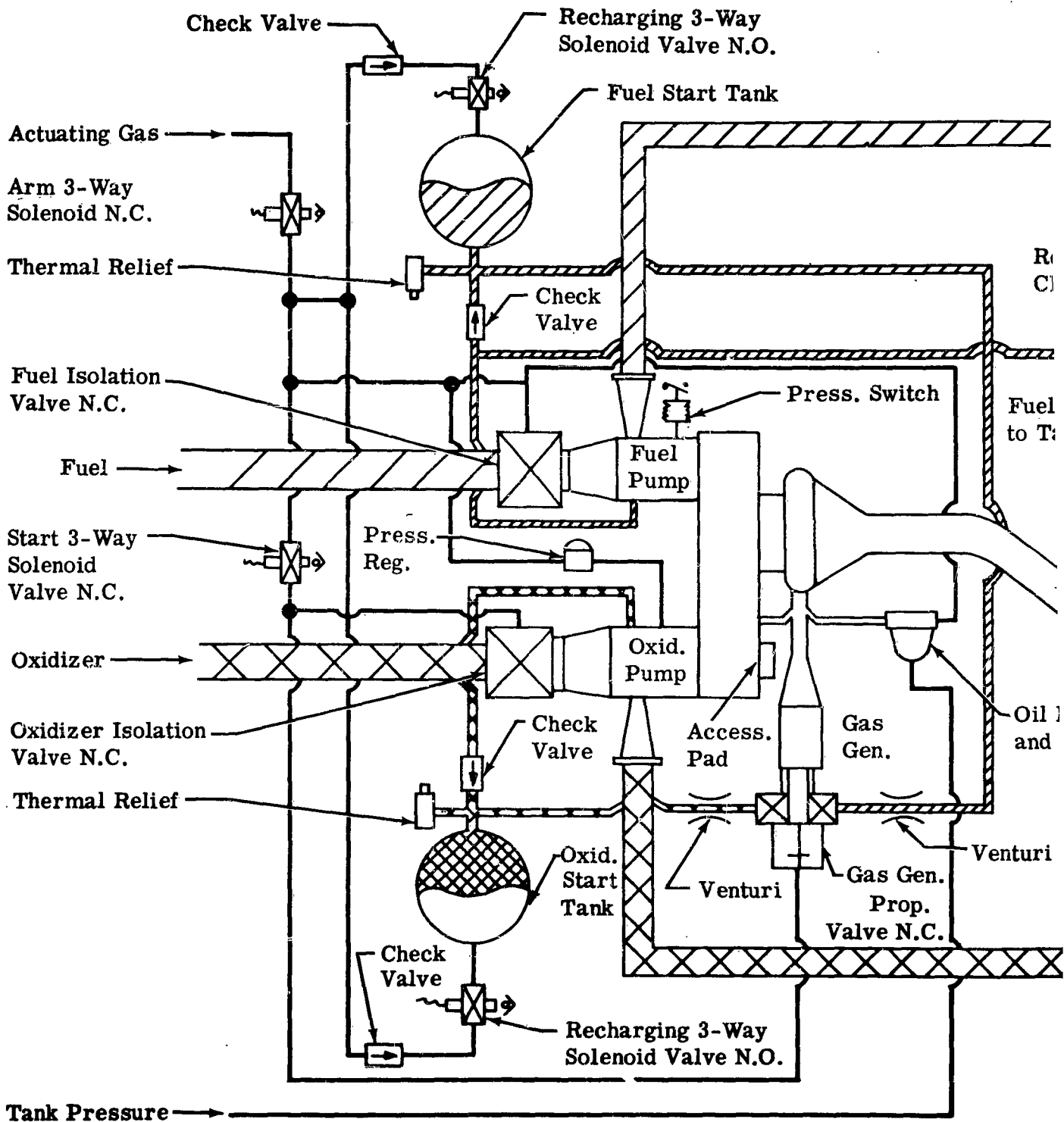


Figure 15. Model 8060 Reaction Control System (Automatic System Only)



1

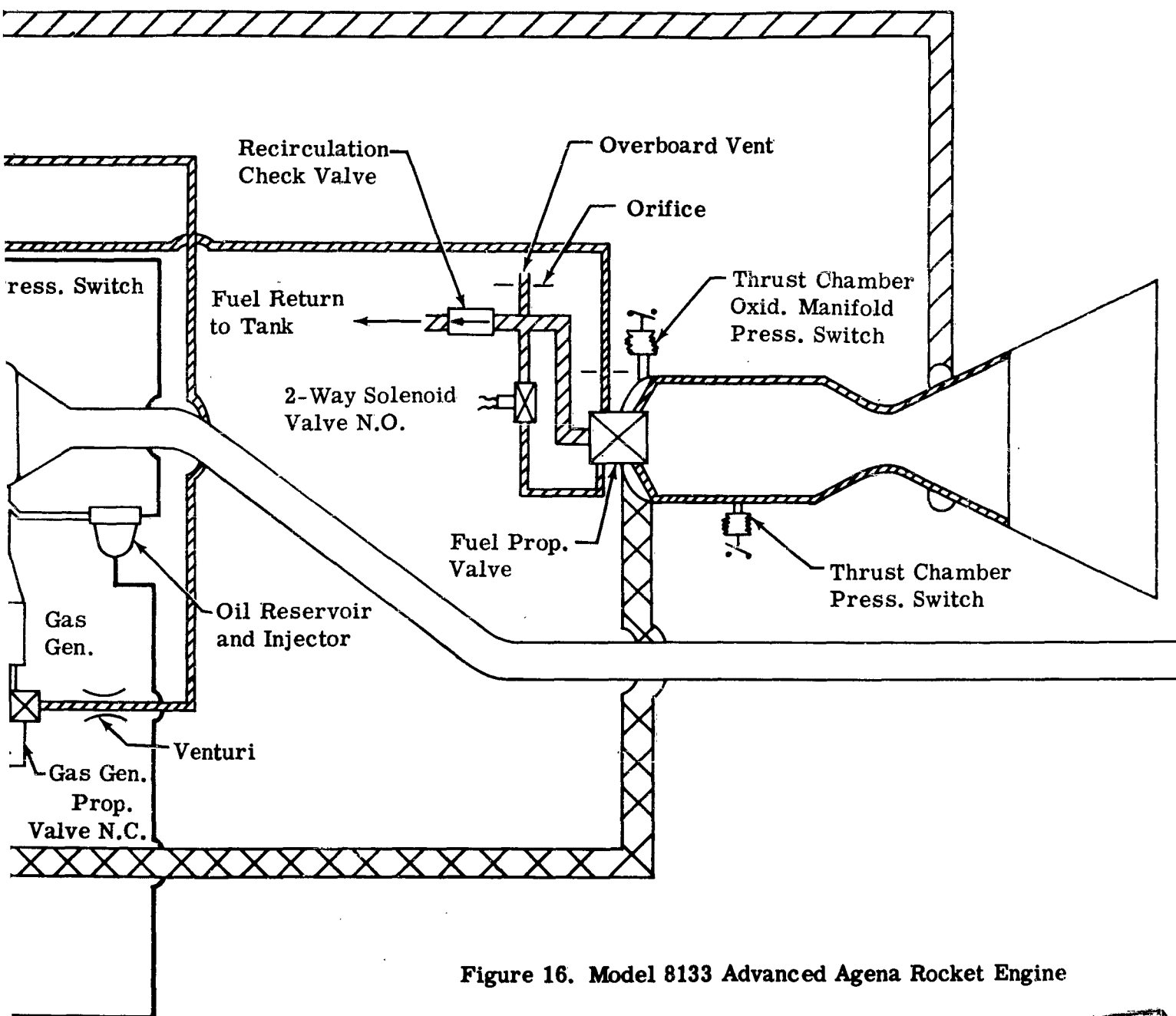


Figure 16. Model 8133 Advanced Agena Rocket Engine

e. Saturn S-IV

The Saturn S-IV stage employs Pratt and Whitney rocket engines designated as the RL10A-3. These engines use liquid oxygen and liquid hydrogen as propellants producing 15,000 pounds of thrust from each of six chambers. Features include a centrifugal turbine pump having three stages, one for the oxygen flow and two for hydrogen, a multiple restart capability and firing duration of ten minutes. The thrust chambers are cooled by passing liquid hydrogen through an internal jacket and ignition of propellants is accomplished by an electric spark in the injector. A unique feature of this engine is that no lubricant is provided for turbine gears which operate dry with cooling provided by an H_2 gas flow. Helium is employed as a propellant tank pressure transfer gas.

The engine operates at a nominal chamber pressure of 300 psia with a nozzle area ratio of 40. The complete engine assembly is gimballed to permit thrust vector control. Figure 17 shows a schematic of the Saturn S-IV engine.

f. J-85-5 Turbo Jet

This air breathing engine is produced by the Flight Propulsion Division of the General Electric Company. It has an eight stage axial compressor, annular combustion chamber, two stage turbine and afterburner with variable-area nozzle. The maximum rating of the J-85-5 with afterburner is 3850 lb at sea level. It is approximately 104 inches in overall length, 20 inches in diameter, and has a dry weight of 325 pounds.

g. Ion Engine

The ion group of electric propulsion systems was found to offer the highest potential for application to the stabilization requirements of the Military Test Space Station. The other basic forms of electric propulsion, namely electro-thermal (arc jet and resistance heated), electromagnetic or plasma systems, and heavy-particle systems were also examined but found to be less attractive for the following reasons:

- (1) Arc jet systems will be limited to specific impulses on the order of 1500 to 2000 sec.
- (2) Erosion of arc jet electrodes and nozzle walls and cooling are problem areas needing extensive research.
- (3) Plasma engines have thus far exhibited very limited life spans due to sputtering, erosion of chamber and nozzles and short endurance of circuit elements.
- (4) Current ion engines have exhibited sufficient reliability and life during laboratory testing to suggest their use for space missions.

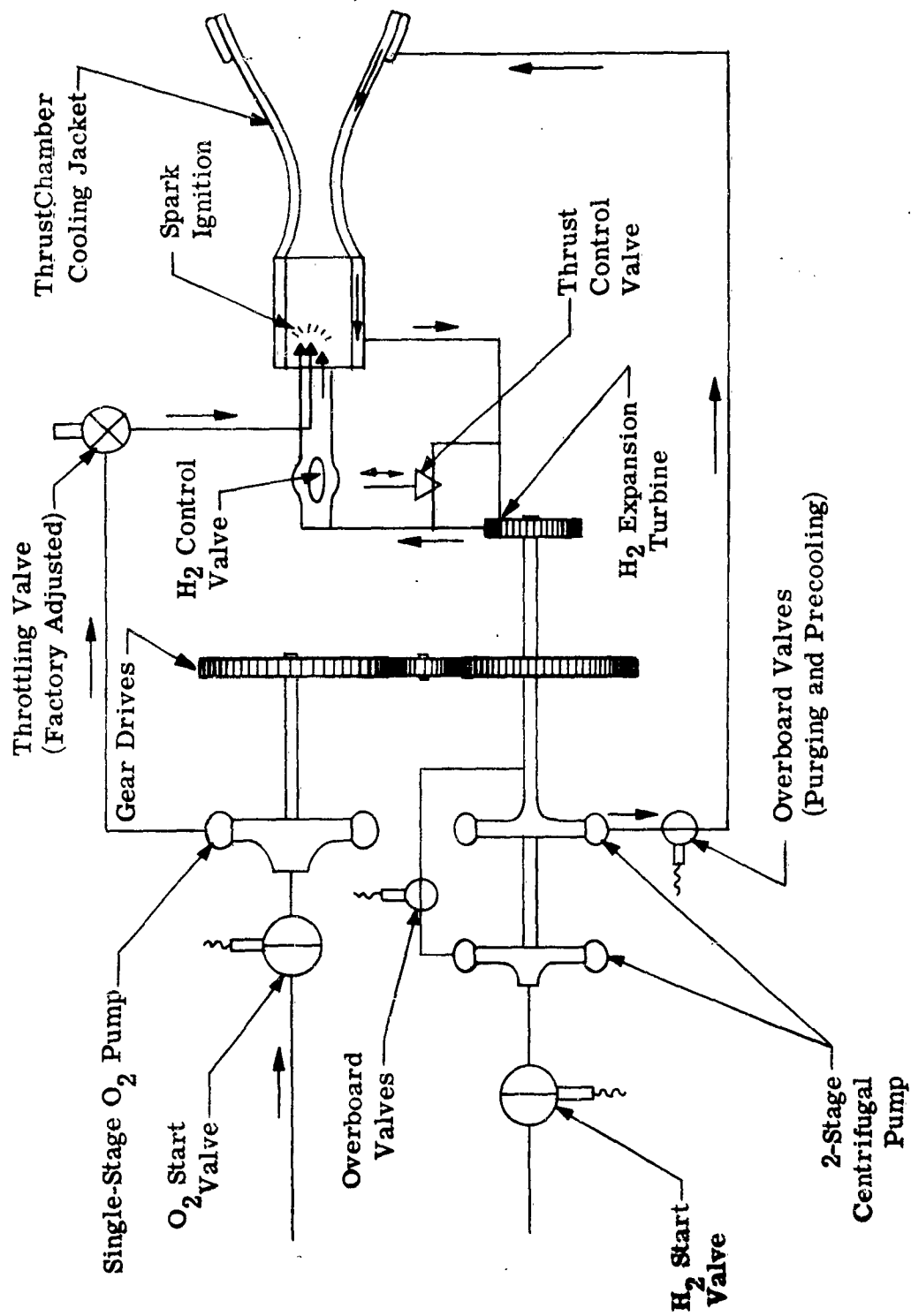


Figure 17. LH₂-LO₂ Rocket Engine

- (5) Ion engine development is further advanced than the other electric propulsion forms due to extensive efforts by various private industries as well as NASA-Lewis Research Center, NASA-Langley, NASA-JPL and NASA-Marshall. Also, joint AF-NASA flight test programs using high altitude rockets are planned for late 1962 and 1963.
- (6) Specific configuration data is available on test proven ion propulsion systems.

An ion engine currently under development for the USAF employs cesium for the ion source since it melts at 83°F and ionizes readily in porous tungsten which is employed as the initiator. To produce a neutral exhaust, one which contains equal amounts of ions and electrons, an accelerating and also a decelerating electrode arrangement is necessary so that the equalizing electrons can be added to the ion stream. Some other significant features of this ion engine include the electrical power source for cesium heaters and electrodes, a mechanically controlled vapor valve for regulating cesium flow, and a neutral "eye" to prevent build-up of a static charge within the space vehicle itself.

2. Maintenance Requirements

In view of the large quantities of components involved in the propulsion systems required by the space vehicles of this contract, it was found desirable to group the various elements into functional categories and develop their maintenance characteristics accordingly. Thus, eleven major component groups were established for the liquid rocket propulsion systems and eight for the ion electric motor. The objective of this grouping was to develop meaningful data relative to propulsion system space maintenance requirements. From a survey of Trouble and Failure Reports (27, 28) received on Bell Aerosystems Company rocket engines which have been fabricated, tested and delivered, it was possible to allocate hard reliability data to the various component groups. Having thus established the types or forms of failure most likely to be encountered, it became feasible to estimate the maintenance requirements of these propulsion systems. Then from reviews of Service and Maintenance Handbooks, and through interviews with working technicians at Bell Rocket Test sites, the necessary repair tasks and sequences were documented. The following data on liquid rocket systems is presented as representative and illustrative of the techniques employed and depth of coverage achieved.

The failures listed in the following analysis were derived from rocket testing experience and from consideration of the vulnerability of the designs to environmental factors and usage. These failures rarely occur on operational systems, however, they provide an insight into the maintainability parameters which should be considered during the design of space vehicles employing similar devices.

a. Major Component or Component Group

(1) Thrust Chambers GFR - N.A.

(a) Failure Type

- (1) Wall burn through causing shutdown or degraded performance.
- (2) Internal coating erosion after successive firings, purge sequences, flushing and pressure tests. Coating loss reduces thermal resistance but is usually not severely critical to performance.
- (3) Welding cracks and/or porosity. Results in degraded performance due to loss of chamber pressure. (This mode is found on low thrust units more often than on the larger engines.)
- (4) Foreign material plugging injector orifices. Results in degraded performance due to loss of chamber pressure. (This mode is found on low thrust units more often than on the larger engines).

(b) Malfunction Cues

Loss in acceleration should be apparent to astronauts through motion cues and/or instrumentation on the vehicle control panel.

(c) Malfunction Verification

Restart or recycle engine and observe performance in terms of acceleration, chamber pressure and/or combustion temperature.

(d) Maintenance Requirement

- (1) Large rocket engines -- if thrust chamber is still operative, the astronauts could accept and make suitable mission adjustments for the degraded performance. An inspection to find exact reason for the loss in thrust would probably be made. Repair and/or replacement of a large thrust chamber appears unlikely with present designs.
- (2) Removal and replacement of small uncooled thrust chamber assemblies is feasible. Calibration requirements may present additional maintenance tasks such as trimming orifice adjustments.

(2) Propellant Valves GFR - N.A.

(a) Failure Type

- (1) Leakages.** This is the most frequent source of trouble. (73 percent of Agena discrepancies observed were due to leaks.) Causes were worn gaskets and seals, inadequate material, and mishandling.
- (2) Poppet jammed open or closed.**
- (3) Failure of poppet springs.**

(b) Malfunction Cues

- (1) Loss of tank pressure and propellants thus preventing a restart.** Decrease in engine operating time.
- (2) Engine may start and not shut down or vice versa.** Restart would be impossible.
- (3) Increase in shut-down time.** Exhaustion of propellants. Restart would be impossible.

(c) Malfunction Verification

- (1) Visual inspection of suspected unit.** A falling off of the propellant tank pressure readings might also give a clue to the malfunction.
- (2) Visual inspection to determine poppet position.** (May not be positive due to small travel (0.070)).
- (3) Valve removal and disassembly for internal inspection.**

(d) Maintenance Requirement (Corrective Action)

- (1) Remove valve, replace with a new unit or replace faulty gasket or seal.** A plumbing fitting could be retorqued to eliminate leakage.

(3) Solenoid Valves GFR - 11.0

(a) Failure Type

- (1) Leakages (see comments on propellant valves)**
- (2) Broken spring**

- (3) Broken wire in solenoid
 - (4) Poppet jammed open or closed
 - (5) Burned out solenoid
- (b) Malfunction Cues
 - (1) Loss of propellants and resulting decrease in engine operating time. Loss of pressurization gas (if a gas solenoid).
 - (2) Valve might operate before coil was energized due to inlet pressure causing unseating.
 - (3) Valve would fail to function upon command.
 - (4) Expected valve functions would not be realized.
- (c) Malfunction Verification
 - (1) Internal vehicle verification would consist of monitoring tank pressure indicators for lowering values. A visual inspection of the suspected unit would be required as a final verification.
 - (2) Valve removal and disassembly for internal inspection.
 - (3) Continuity check of solenoid coil would verify damage to coil wires.
- (d) Maintenance Requirement
 - (1) Same as propellant valves.
 - (2) Remove and replace complete valve.
- (4) Turbine Pump GFR - 14.0
 - (a) Failure Type
 - (1) Leakages — propellant and/or lubricating oil
 - (2) Failure of critical rotating components such as rotors, shafts, inducer, gears, impeller and oil slingers.
 - (3) Weld failure
 - (4) Bearing failure
 - (5) Alteration of pump performance characteristics. This could be caused by distortion of the inducer profile, loss of head rise or flow, roughened internal surface finish.

(b) Malfunction Cues

- (1) Loss of propellants and decrease in engine operating time. Explosions and/or fire could occur if propellants are hypergolic and both fuel and oxidizer leakages develop. Turbine shaft seal leakage could permit propellant entry into gear cases resulting in fire and/or explosions. Lube-oil leakages could cause gear failure and functional shutdown of engine.**
- (2) Engine shutdown.**
- (3) Increased torque, temperature, and wear eventually causing hardware failure and engine shutdown.**
- (4) Thrust level degradation**

(c) Malfunction Verification

- (1) Small leakages could only be verified by actual part inspection which might require disassembly. Gross external leakages could be verified visually. Loss of propellants might also be verified by monitoring turbine pump pressure indicators for out of normal indicators.**
- (2) Visual inspection of disassembled turbine pump parts.**
- (3) Monitoring engine performance in terms of acceleration and/or chamber pressure.**

(d) Maintenance Requirement

- (1) The maintenance requirements for failure types 2 - 5 would indicate that the complete turbine pump must be removed and replaced. This is an involved, time consuming and exacting task even in a terrestrial environment. Most pumps are located just forward of the thrust chamber injectors and are thus inaccessible. Minor leakages (type 1) and minor performance degradation would probably be tolerated if mission accomplishment would not be affected. Minor repairs of leaking fittings is feasible.**

(5) Gas Generators (Ballistic, mono- or bi-propellant types) GFR - N.A.

(a) Failure Type

- (1) Leakages (gas or liquid)**
- (2) Weld failure**

- (3) Broken wire in solenoid
 - (4) Poppet jammed open or closed
 - (5) Burned out solenoid
- (b) Malfunction Cues
- (1) Loss of propellants and resulting decrease in engine operating time. Loss of pressurization gas (if a gas solenoid).
 - (2) Valve might operate before coil was energized due to inlet pressure causing unseating.
 - (3) Valve would fail to function upon command.
 - (4) Expected valve functions would not be realized.
- (c) Malfunction Verification
- (1) Internal vehicle verification would consist of monitoring tank pressure indicators for lowering values. A visual inspection of the suspected unit would be required as a final verification.
 - (2) Valve removal and disassembly for internal inspection.
 - (3) Continuity check of solenoid coil would verify damage to coil wires.
- (d) Maintenance Requirement
- (1) Same as propellant valves.
 - (2) Remove and replace complete valve.
- (4) Turbine Pump GFR - 14.0
- (a) Failure Type
- (1) Leakages — propellant and/or lubricating oil
 - (2) Failure of critical rotating components such as rotors, shafts, inducer, gears, impeller and oil slingers.
 - (3) Weld failure
 - (4) Bearing failure
 - (5) Alteration of pump performance characteristics. This could be caused by distortion of the inducer profile, loss of head rise or flow, roughened internal surface finish.

(b) Malfunction Cues

- (1) Loss of propellants and decrease in engine operating time. Explosions and/or fire could occur if propellants are hypergolic and both fuel and oxidizer leakages develop. Turbine shaft seal leakage could permit propellant entry into gear cases resulting in fire and/or explosions. Lube-oil leakages could cause gear failure and functional shutdown of engine.**
- (2) Engine shutdown.**
- (3) Increased torque, temperature, and wear eventually causing hardware failure and engine shutdown.**
- (4) Thrust level degradation**

(c) Malfunction Verification

- (1) Small leakages could only be verified by actual part inspection which might require disassembly. Gross external leakages could be verified visually. Loss of propellants might also be verified by monitoring turbine pump pressure indicators for out of normal indicators.**
- (2) Visual inspection of disassembled turbine pump parts.**
- (3) Monitoring engine performance in terms of acceleration and/or chamber pressure.**

(d) Maintenance Requirement

- (1) The maintenance requirements for failure types 2 - 5 would indicate that the complete turbine pump must be removed and replaced. This is an involved, time consuming and exacting task even in a terrestrial environment. Most pumps are located just forward of the thrust chamber injectors and are thus inaccessible. Minor leakages (type 1) and minor performance degradation would probably be tolerated if mission accomplishment would not be affected. Minor repairs of leaking fittings is feasible.**

(5) Gas Generators (Ballistic, mono- or bi-propellant types) GFR - N.A.

(a) Failure Type

- (1) Leakages (gas or liquid)**
- (2) Weld failure**

- (3) Clogging of filters in gas feed lines
- (4) Gas generator igniter failure
- (5) Plugged injector orifices (liquid type gas generator)

(b) Malfunction Cues

- (1) Reduced speed of turbine pump and thrust degradation.
- (2) Engine would not start. (Redundant igniters are usually provided.)
- (3) Slight pump performance shifts and/or local high temperatures.

(c) Malfunction Verification

- (1) Verification would be difficult since leaks occur only during engine operation.
- (2) Monitoring pump performance could provide a clue to pressure losses caused by weld porosity or other leakages.
- (3) Verify igniter signal continuity (probably external to the vehicle).

(d) Maintenance Requirement

- (1) Joint or fitting leakages could be repaired by installing new seals or gaskets or by retorquing. Some pump performance degradation might be tolerated if mission compromise would not result.
- (2) Gross weld failure would require complete component replacement.
- (3) Filters should be accessible for cleaning and/or replacement.
- (4) Failure to ignite would require electrical continuity checks, igniter cleaning or replacement, or if a solid charge is used removal and replacement of the grain.
- (5) Injector should be accessible for inspection and cleaning.

(6) Filters GFR - 0.3

(a) Failure Type

- (1) Flow degradation**
- (2) Filter blow-outs would permit passage of foreign particles to the critical areas.**

(b) Malfunction Cues

- (1) Performance degradation (i.e., pump, thrust chamber)**
- (2) Termination of operation. Plugging of injector and consequent overheating, component failures due to injection of foreign matter.**

(c) Malfunction Verification

Performance monitoring would be possible clue. Visual inspection would be required for positive verification.

(d) Maintenance Requirement

All filters should be located to permit removal, inspection and replacement.

(7) Check Valves GFR - 5.0

(a) Failure Type

- (1) Leakages (external, internal)**
- (2) Broken poppet spring or jammed open or closed.**

(b) Malfunction Cues

- (1) Loss of propellants or gases and resulting decrease in engine operating times.**
- (2) Internal leakages, or failure of the valve to restrict reverse flow could cause mixing of hypergolic propellant gases and explosive conditions.**

(c) Malfunction Verification

- (1) External leaks would be verified by visual inspection.**
- (2) Internal leaks would require valve removal for disassembly and replacement.**

(d) Maintenance Requirement

- (1) Correction of minor external leaks would require replacement of seals or 'O' rings or retorquing.**
- (2) Remove and replace valve.**

(8) Pressure Regulators GFR 2.14

(a) Failure Type

- (1) Leakages (external or internal)**
- (2) Broken spring, jammed open or closed poppet, or ruptured diaphragm.**

(b) Malfunction Cues

- (1) Loss of pressurization gas and resulting decrease in propellant utilization rate.**
- (2) May cause a gradual build-up of gas pressure down stream of regulator and eventual damage to sensitive components, (i.e., tanks, bladders). Some regulators have relief protection features to eliminate this occurrence.**
- (3) A jammed closed regulator could prevent gas pressure from flowing downstream thus engine starts or gas operated valve functions would be impossible.**

(c) Malfunction Verification

- (1) Visual inspection.**
- (2) Remove and disassemble regulator for inspection, repair (if possible) and replacement.**
- (3) Indications of regulator malfunctions might be observed if the system includes pressure transducers feeding information to internal gages. Monitoring these gage values against established standards would verify regulated pressures which are out of tolerance.**

(d) Maintenance Requirements

- (1) Remove and replace or repair regulator.**
- (2) If observation of pressure indicators showed minor discrepancies, adjustments could be made to correct. These**

adjustments consist of changing the diaphragm spring rate of the regulator by turning an adjustment screw which varies the spring height.

(9) Plumbing GFR - 0.1 (for fittings)

(a) Failure Type

Leakages caused by worn gaskets and seals, poor workmanship and mishandling. Pipe thread leaks can also occur when improper sealants are applied.

(b) Malfunction Cues

Loss of propellants or gases and resulting decrease in duration of functions.

(c) Malfunction Verification

Inspection and leak test externally. Unusual tank pressure gage readings would also provide a clue.

(d) Maintenance Requirement

Retorque or remove and replace leaking unit. Damage to fitting or tube sealing surfaces would require complete item removal and replacement.

(10) Propellant Tanks (Including gas pressurization tanks) GFR - 0.15 for general tanks; 0.07 for helium storage tanks

(a) Failure Type

- (1) Leakages.** Helium gas leaks are often found in the vicinity of welds or fitting bosses. Fluid leaks are not as prevalent. Bladder expulsion devices are also leak prone due to workmanship, improper installation or mishandling.
- (2) Structural tank failures** could occur due to excessive internal pressure. This might develop through decomposition of propellants, regulator failure or excessive vapor pressure.

(b) Malfunction Cues

- (1) Loss of propellants or gases as indicated by tank pressure gages.** Reduced duration of engine thrust.

(2) Explosions could occur as a result of gross tank rupture.

(c) **Malfunction Verification**

Minute gas leaks are isolated normally with soap solutions or with a leak detector instrument. Gross leaks can be seen visually or audibly noted.

(d) **Maintenance Requirement**

Tank fitting bosses use gaskets, 'O' rings and jam nuts for sealing. Threads are also coated with a sealant for the same purpose. Replacing gaskets, 'O' rings and sealants might correct the leak.

(11) **Electrical Cordages and Components** GFR - 0.02 for cable assemblies; 0.2125 per pin for AN type connectors

(a) **Failure Type**

- (1) Damaged connector pins
- (2) Connector separation
- (3) Improper pin contact
- (4) Insulation damage or breakdown
- (5) Insert cracks
- (6) Shorts
- (7) Seal deterioration
- (8) Relay contacts fail to actuate (close or open or function momentarily)
- (9) Shorted, open or out of tolerance resistor(s)

(b) **Malfunction Cues**

- (1) Partial or complete functional breakdown. Expected events do not occur (i.e., valve operations, engine starts, transducer feedback, etc.).
- (2) Shorts and flash overs will develop. This may cause intermittent or unusual equipment operation.
- (3) Occasionally, normal operation will occur since redundancies are usually provided for critical circuits (i.e., engine ignition).

(c) Malfunction Verification

- (1) Visual inspection of a separated connector.
- (2) Functional checkout would be most feasible verification task. Most relays are located inside a relay box which is completely encapsulated thus making it impossible to inspect internal resistors, relays or contacts.

(d) Maintenance Requirement

- (1) In the case of connectors having only a few wires, removal and replacement of the connector may be feasible. Connectors having many wires would require removal of the complete cordage (wires and connector) to correct damage or malfunction. Spare pins are often provided and could be used to correct a damage. Continuity check-out would be required to isolate the faulty pin or wire.
- (2) Most relay boxes are sealed units, thus the only possible repair would be removal and replacement.

G. LIFE SUPPORT SYSTEMS

The life support systems selected for analysis reflect the current state-of-the-art approaches and components. In general, life support systems fall into two categories; first, those devices needed for management of compartment oxygen, carbon dioxide, pressure, temperature, humidity, odors, noxious gases, sanitation, hygiene and waste products. Secondly, those items needed for a completely closed ecological system which would provide, not only the respiratory requirements of the space cabin, but also the nutrients required by the crew (e.g., an algae life support system). Reference 29 presents a discussion of the experimental program in progress at the USAF School of Aviation Medicine relative to closed ecological systems employing an algal cell photosynthetic approach. It should be noted, however, that, even with the encouraging results obtained by these studies, the development status is still exploratory. In view of these facts it was decided to postpone to a later time, development of the maintenance requirements of a completely closed ecological system and concentrate on more conventional systems.

1. System Concepts

a. Rotational Open Cycle Systems

These systems are composed of cryogenic supplies of oxygen and nitrogen which, after gaseous conversions, are routed through pressure reducing and regulating valves and then into a mixer which properly proportions the gases for crew consumption and compartment pressurization. Space suit connections are also provided for emergency provisions. In the large space stations such as ESWS and MTSS a water reclamation and purification system is also desirable. In addition, it becomes necessary to remove carbon dioxide, particulate matter, odors and noxious gases from the breathing system. Provisions for both water reclamation and gas purification can be incorporated in a single unit made up of a debris filter, an activated carbon odor removal filter, a lithium hydroxide CO_2 absorber, a water condenser/separator and a water collector. This device is usually integrated into the cabin return air duct together with fans and motors for flow control and status indicators showing temperature, CO_2 and partial pressures.

b. Integrated Thermal and Atmospheric System

The integrated approach combines the two basic requirements of thermal and atmospheric control into a single system. A typical space station configuration employing this concept would consist of an air circulating network of ducts routed to appropriate sections of the vehicle with the return elements connected to an air processing unit. This unit consists of an intake grill and electrostatic filter which removes dust, smoke and other foreign material, and a radiator panel which automatically maintains a preselected thermal balance. Since variations

in cooling loads are to be encountered a bimetallic strip is employed which controls the radiator area being utilized. Humidity control is accomplished by condensation of water vapor from the air to the radiator surface where it is picked up by the air flow carried to a centrifugal water-air separator and stored in a tank for use by the crew.

Purification in this system is accomplished by passing the air through an H_2 flame and an activated charcoal bed which removes CO_2 , CO, H_2 and noxious gases. If desirable, oxygen can be added at this point by passing the air through a potassium superoxide bed. Most configurations recommend a high pressure or cryogenic oxygen system as a backup measure to the KO_2 liberation bed. Movement of air in the system is provided by a variable speed heavy duty circulating fan and motor.

The crew complement of ESWS and MTSS may consist of as many as fifteen men who require about 5.7 pounds of water per man-day, and produce, in the form of urine, about 4 pounds per man-day. In view of these quantities, it becomes desirable to recover potable water from this source. This can be accomplished by employing a centrifugal boiler which distills urine collected from the crew, removes odors in an activated charcoal bed and recondenses the product for storage and later use.

c. Closed Loop Environmental System

The Mercury System, which is representative of this approach, provides the cabin and space suit with a pressure of 5.1 psia and an atmosphere of 100 pounds oxygen. A common coolant water supply and electrical supply is provided for suit as well as cabin use. Flow of water from this supply to the heat exchanger is provided by a pressurized bladder. Oxygen, in the gaseous state, is stored in spherical tanks at 7500 psi.

Two pressure suit connections are provided, one at the helmet for exhaust and one at the waist for inlet. The outlet connection carrying metabolic oxygen, CO_2 , water vapor, and debris passes from the suit to a debris trap and into an odor absorber bed of activated charcoal and lithium hydroxide. Cooling of the gas takes place in a water evaporate type heat exchanger which utilizes the space vacuum to cause coolant water to boil at approximately $35^\circ F$. The steam produced is dumped overboard. The gas side of this heat exchanger picks up water vapors from the suit and condenses them into droplets which are carried to a mechanical water separator and eventually into a small tank. Condensed water is caught in a vinyl sponge which is squeezed out periodically into a container. A constant bleed orifice was provided between the O_2 supply and the pressure suit in order to supply a continuous flushing action, as well as adequate partial pressure. Should cabin decompression occur, the demand regulator automatically establishes a reference pressure of 4.6 psia for the exhaust port of the regulator, thereby maintaining suit pressure.

Two gaseous oxygen tanks are provided giving supplies sufficient for 56 hours of operation. These tanks are equipped with pressure transducers to provide pressure data indication to the occupant.

Cabin pressure is controlled by a relief valve which maintains a differential pressure of 5.5 psi during vehicle climb. A manual decompression device is incorporated to permit dumping of cabin pressure in emergency conditions. A cabin pressure regulator meters O_2 to maintain the lower limit of pressurization at 5.1 psia. A manual recompression device is also provided on this regulator. Cabin temperature is maintained by a fan and heat exchanger similar to that employed for the suit.

A closed loop system for use on the backpack of an extravehicular space suit would employ similar components for debris, odor and CO_2 removal, O_2 supply, and circulation and pressure management. In addition, a method of supplying life-support capabilities for two men from one source should be incorporated since team functions and rescue operations may be required.

d. Typical Environmental System

The previous sections have treated generalized approaches to life support systems. However, to enhance the maintenance analysis which follows, it was deemed desirable to present details of a reasonably complete environmental control system. The schematic of Figure 18 illustrated a system meeting these requirements which could be employed for the lunar vehicle described in Chapter 2. CO_2 produced by the vehicle crew is removed in this schematic by reacting with anhydrous lithium hydroxide forming lithium carbonate. This reaction is not reversible, thus the $LiOH$ becomes saturated and eventually spent during the mission. A minimum of approximately 3 1/2 pounds of $LiOH$ are required by each crew member each day of the mission. For an added safety margin, the Mercury capsule employed 5.1 pounds of absorbent. Other CO_2 removal materials include potassium, lithium and sodium superoxides which can also be employed as a source of oxygen for the crew of the station with a gaseous oxygen system for backup - (3,000 pounds of KO_2 liberates 1,000 pounds of O_2).

The primary oxygen supply system consists of stored super critical cryogenic gas. Supplies are sufficient for 50 percent excess required for normal metabolic and leakage needs plus two complete cabin repressurizations and a minimum of 18 air lock operations. Resupply of space suit back-pack life support systems is also provided.

The space radiator shown is integral with the vehicle skin and susceptible to meteor damage. Thus, redundant coolant loops are recommended.

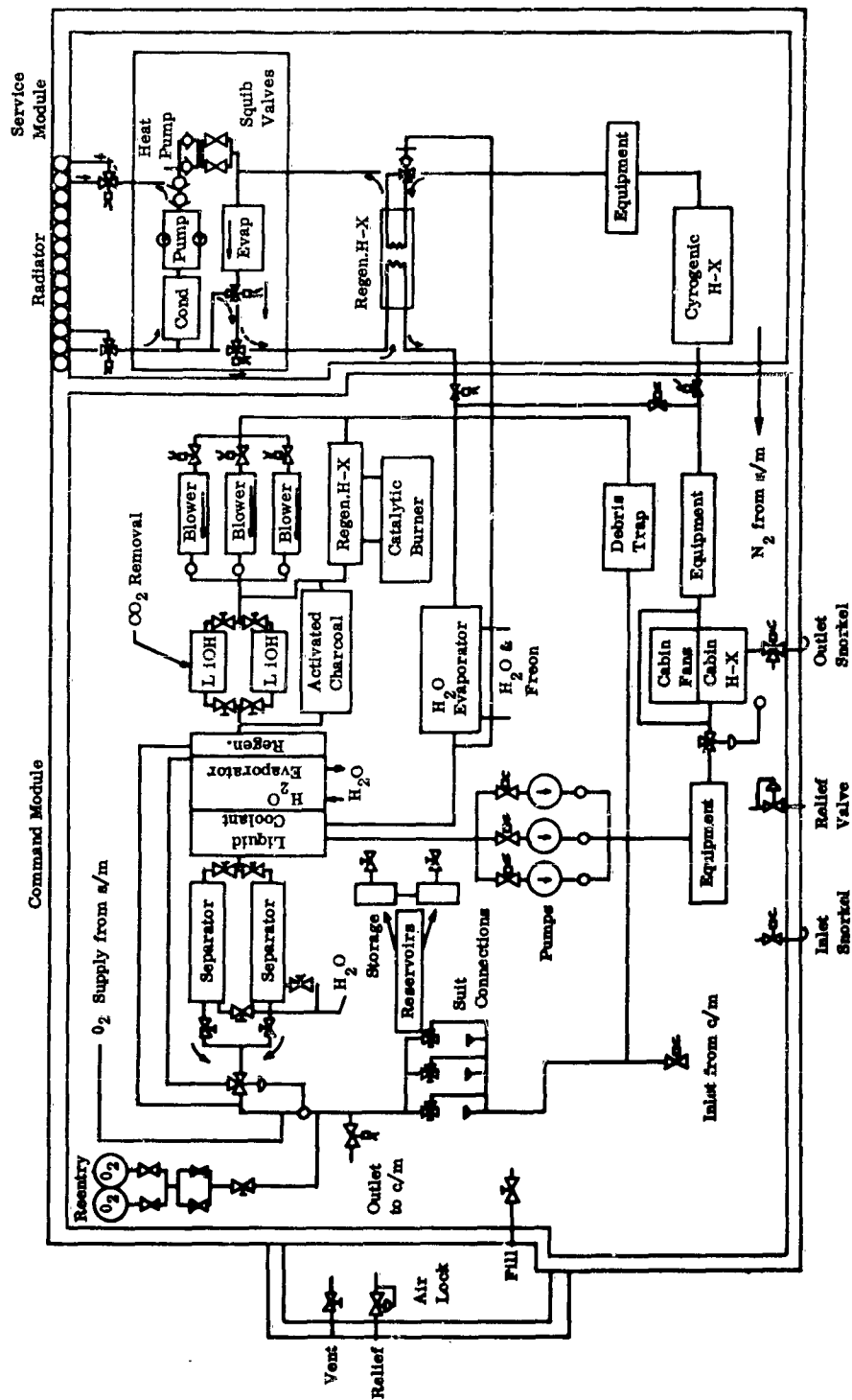


Figure 18. Typical Environmental Control System

2. Maintenance Analysis Life Support Systems

This analysis will employ the functional grouping technique first illustrated in the Section on Space Vehicle Propulsion. The data presented are considered representative of the technique used, and depth of coverage achieved, during this phase of study.

Major Component or Component Group

a. Valves

This group includes solenoid on-off, relief, vent, fill, check, squib and selector types. GFR-8.7 average.

(1) Failure Type

- (a) Leakages caused by worn or improperly installed gaskets, 'o' rings or seals, inadequate materials, poor workmanship or mishandling.**
- (b) Jammed poppets or spools.**
- (c) Broken, or damaged actuating springs.**
- (d) Broken wires within the solenoid or burned out unit.**

(2) Malfunction Cues

- (a) Loss of fluids or gases causing malfunctions and/or degraded performance. Cabin instruments and indicator lights would give warnings of hazardous conditions.**
- (b) Valve functions would not occur upon demand and system malfunction would result.**
- (c) Valve function might occur without being programmed. Check valves would not limit flow as intended causing system malfunctions.**
- (d) Valve would fail to function upon demand.**

(3) Malfunction Verification

- (a) Immediate verification could be obtained by monitoring system pressure indicators for decaying values. Exact location could be deduced from malfunction lights and gage indications. Some malfunctions can be corrected from flight stations by the crew. This task might involve selection of the emergency mode in a redundant system, changing system flow characteristics by proper valve movements, or by system shutdown and switchover to a pressure suit environment. Visual inspection of the suspected unit would be the final verification.
- (b) Actual isolation of the malfunction might require removal and disassembly of the valve.
- (c) Continuity check of solenoid coil would verify damage to wires.

(4) Maintenance Requirement

Remove valve, replace with a new unit or replace faulty gasket, 'O' ring or seal. Retorquing of valve nuts or bolts might correct some leakages.

b. Pumps GFR 13.5

(1) Failure Type

- (a) Seal leakages
- (b) Internal damage

(2) Malfunction Cues

- (a) System pressure drops or does not build up.
- (b) System fluid temperature becomes excessively high. Pressure surges would also be a clue to pump internal failure.

(3) Malfunction Verification

- (a) Monitor appropriate pressure indicator in the cabin. Inspect pump shaft seals for signs of fluid leakage, inspect for signs of external damage.

- (b) Check seals for leakage, if none is found, damage must be internal.

(4) Maintenance Requirement

- (a) Remove and replace faulty seals or retorquer leaking fittings. Pump may require replacement if leakage is serious.
- (b) Replace pump. Some current pumps are designed with replacement cartridges for high wear parts. These can be installed without pump removal.

c.. Electric AC Motors GFR-Fans .225, Motors .625, Blowers 2.5, Blower Motors 0.2

(1) Failure Type

- (a) Brinelling of bearings.
- (b) Shorts, opens, deterioration of lubricants.
- (c) Binding of bearings and corrosion.
- (d) Oxidation.

(2) Malfunction Cues

- (a) Stoppage or reduction of circulation within the environmental control system. Since blowers are employed to cause flow of air through the CO₂ removal beds and the odor removal cannister, any reduction in fan velocity could create a dangerous compartment atmosphere. CO₂ and O₂ pressure indicators would give clues to fan or blower malfunctions.

(3) Malfunction Verification

- (a) Critical situation gages and indicator lights would reveal a malfunction within the system.
- (b) Visually inspect unit for damage or obstructions which might cause failure.

(4) Maintenance Requirements

- (a) Switch over to emergency mode of system operation (space suit dependency).

(b) Remove and replace motor and fan.

d. Heat Exchangers GFR - 15.0

(Regenerative and cryogenic)

Includes:

**Evaporators
Radiators
Condensers (fluid)
Separators**

(1) Failure Type

(a) Leakages caused by external damage, corrosion, fittings or seal deterioration, weld porosity, excessive thermal stresses, fabrication defects, meteoroid penetration and erosion.

(b) Accumulation of scale, sludge and/or slime.

(2) Malfunction Cues

(a) Performance degradation, pressure drops and eventual system failure. (Changes in thermal characteristics.)

(b) Reduction in heat transfer properties.

(3) Malfunction Verification

Monitoring of appropriate gages would give indications of system performance degradation. Visual inspection, both internally and externally would be required for gross leak detection.

(4) Maintenance Requirement

(a) A decision to accept performance degradation based on system safety margins could be made. Plumbing fittings could be replaced or torqued up if a leak is isolated. Leaks in surface mounted radiators could be repaired. Severely damaged components could be isolated in the system and/or removed and replaced (if logistically feasible.)

(b) Radiators may be coated to improve emissivity. This coating may be gradually eroded away by meteoroids, thus changing the radiator thermal characteristics.

e. CO₂ Removal bed. GFR N.A.

(1) Failure Type

Saturation of the absorber with CO₂.

(2) Malfunction Cues

Partial pressure of CO₂ in cabin or suit increases.

(3) Malfunction Verification

Since operation of the CO₂ removal bed is time sensitive, a check of the operating interval for a particular cannister would serve to verify saturation of that bed.

(4) Maintenance Requirement

Remove and replace CO₂ removal bed.

f. Odor Removal Bed GFR-N.A.

(1) Failure Type

Saturation of the absorber.

(2) Malfunction Cues

Odors increase.

(3) Malfunction Verification

Since operation of the odor removal bed is time sensitive, a check of the elapsed operating time would verify bed saturation.

(4) Maintenance Requirement

Remove and replace odor removal cannister. Activated charcoal can be rejuvenated by applying heat and exposure to vacuum.

H. SPACE VEHICLE POWER SUPPLIES

The various elements making up the Military Test Space Station exhibit electric power supply requirements covering nearly every concept of the technology. The prime power source for the station proper, for example, must supply an average of 100 kw with a peak demand of about 300 kw. In view of the extended life expectancy for this demand, a nuclear reactor is deemed the most practical solution. A back-up emergency power supply system is also included which consists of a piston engine and alternator set together with a coolant system, voltage and frequency controls, and a switching network. The engine employs hydrazine and nitrogen tetroxide, the same propellants as used by the MTSS. Since the surface area of the station is some 25,000 square feet, a solar cell bank and silver or nickel-cadmium storage battery system could also be employed.

The logistic vehicle required for crew rotation and resupply of the MTSS has a much less exotic power requirement. A chemical turbo electric generator and storage battery combination fulfills the requirements of this vehicle adequately.

The space shuttle employed for assembly of MTSS has short term power requirements similar to those of the logistic vehicle. The one man shuttle described in Chapter II incorporates a chemical energy conversion fuel cell power supply to produce 7.5 kw normal demand amounts. Other systems considered include a closed cycle heat engine, solar voltaic cells and zinc-silver oxide batteries.

The Space Suits provided crewmen on MTSS will derive their power from either the mother ship via a connecting cable, by a battery carried on the suit, by a self-contained power generator such as a fuel cell, or from a beamed source aboard MTSS. Lightweight silver cadmium batteries were considered most convenient and reliable.

To summarize then, the electrical power supplies required by the Military Test Space Station system include:

- (1) Nuclear-turbo electric
- (2) Piston engine and alternator
- (3) Photo voltaic cells
- (4) Chemical turbo-electric
- (5) Fuel cell
- (6) Dynamic heat engine
- (7) Storage batteries

These seven methods for providing electric power to space vehicles of various sorts are representative of those analyzed for maintenance requirements during the contract. Thus, a brief description of the techniques involved and components provided is included below:

1. Nuclear-Turboelectric

Most development in this field has been concentrated in the SNAP and SPUR programs. Various levels of power output from the 300 watt SNAP-2 to the 1 MW SNAP-50 system are currently under contract. (54)

The SNAP-2 system which is fairly typical of the family employs a zirconium hydride reactor fueled with fully enriched uranium and cooled by liquid sodium-potassium alloy. The heat energy available in the reactor will be converted to electricity by a mercury-vapor Rankine-cycle unit. The mercury vapor is condensed in a radiator after expansion through a mechanical turbine. A close loop operation is maintained by returning the mercury condensate to the storage boiler. The mechanical power output of the mercury turbine is converted to electric power by an alternator. The principal components of the SNAP-2 system include the reactor and its mechanical controls, the liquid metal pump, mercury boiler, vapor turbine, mercury vapor radiator/condenser, the mercury feed pump, and the alternator.

2. Piston Engine

The piston engine employed as a secondary power supply system for MTSS utilizes hydrazine and nitrogen tetroxide in a modified two-stroke Otto cycle. In view of the similarity between this engine and small, high performance gasoline engines currently in use a detailed description will not be given. The most significant components in addition to the motor-generator include the electrical controls, alternator and cooling system, and propellant storage and delivery network.

3. Photovoltaic Cells

This system converts solar radiation to electric power by employing a silicon, or like, cell having the capability of transforming light energy to electrical energy. An array of these cells are interconnected to provide a voltage sufficient for charging the vehicle batteries. Blocking diodes are connected in series with the cells to prevent reverse current flow from the storage batteries.

The solar cell modules employed in the Able-4 paddles are representative of current techniques. (53) The basic unit consists of a strip of five boron-diffused cells soldered into shingles. Ten strips are connected in series on the anodized outer surfaces of a tapered block of honeycomb aluminum to form a module. Epoxy glue is used for cell mounting. The modules are attached to a central tube with 11 modules extending to each side. The tube extends beyond the last module for attachment to arms on the payload. Modular replacement is accomplished if faulty cells are encountered after assembly. The storage batteries used consisted of two packs of 14 nickel-cadmium cells each, connected in series.

4. Chemical-Turboelectric

Chemical-turboelectric conversion systems generate electrical power by utilizing turbines driven by the combustion of chemical fuels. Various mono-propellants such as hydrogen peroxide, ethylene oxide, and hydrazine, and combinations of bipropellants such as hydrogen and oxygen are available as sources of energy. Components include a fuel source, flow control valves, propellant feed subsystems, gas generator, turbine, a battery and an electrical conversion unit consisting of an alternator, rectifier and static inverter/converter.

5. Fuel Cell

The fuel cell is a device which converts chemical energy to electrical energy. It is essentially a primary battery in which the fuel and the oxidizer are stored outside the battery and fed to the cell when electrical output is desired. A simplified cell contains two electrodes, an anode and cathode, and the electrolyte which acts as a medium for the transport of charges within the cell. The electrodes act as mechanical devices to bring the reactants in contact with the electrolyte in a controlled way. They act as catalysts or as catalyst carriers to promote the electro-chemical reaction and serve to carry the current generated by the electrochemical reaction to the load. The anode is the electrode at which the fuel gives up electrons to be delivered to the load. The cathode is the electrode which gives up electrons to the oxidizers.

Fuel cells are combined into series and/or parallel arrays for high voltage or high current applications. They may be classified as hydrogen-oxygen molten salt electrolyte, Redox (reduction-oxidation), regenerative, consumable electrode and special types.

The hydrogen and oxygen cell developed by Union Carbide Consumers Products Company is typical of the art. (55) It operates at 60 to 80°C and near atmospheric pressure in an electrolyte of 30 to 35 percent solution of potassium hydroxide in water. The electrodes are of porous carbon on which catalytically active material has been deposited and wet proofed to prevent electrolyte from entering the fine pores and blocking the diffusion of gas. The catalysts which work best on the hydrogen electrode are metals of the platinum group. The water formed in the reaction diffuses back into the gas space where it is removed by passing the wet fuel gas through a loop containing a condenser. Union Carbide cells have been operated for two years in the laboratory at 50 amperes/ft² with little loss in performance. Electrodes of about 4 square feet in area have been fabricated and found to operate satisfactorily.

6. Dynamic Heat Engine

The dynamic heat engine, sometimes called a solar-thermionic converter, utilizes electrons as the working medium. A cathode is heated, for example, by a solar energy concentrator thereby emitting electrons which travel through inter-electrode space to the anode. (56) The circuit is completed by electron flow from the anode to the output load. Current heat engines utilize the ions produced by a capsule of cesium mounted in conjunction with the cell anode.

To obtain the high temperatures required for operation, a solar energy concentrator is utilized. Orientation of the concentrator toward the sun must be maintained with great accuracy to be assured of efficient operation.

7. Maintenance Analysis Power Supplies

The diversity of power supply requirements exhibited by the various elements of The Military Test Space Station suggests using this system to illustrate the methodology employed for development of the various failure types and maintenance requirements.

Nuclear-turboelectric

a. Failure Type

- (1) Beryllium drum reflector control linkage failure, coolant leakage, corrosion by liquid metal.
- (2) Turbine failure due to creep properties of metals in the space environment, corrosion by liquid metal.
- (3) Liquid metal corrosion of radiator tube, boiler, piping and pump.
- (4) Radiator leakages due to meteorite impact

b. Maintenance Requirements

- (1) At this time, it is considered advisable to remotely eject the nuclear reactor when serious malfunctions occur. A "safe" replacement can then be installed.
- (2) The electrical conversion equipment should be located in a radiation free area to facilitate maintenance.

- (3) Replacement of units such as turbine assembly, boiler, heat capacitor, and radiator tubes is considered feasible. Replacement of the units in the space environment requires allowing the liquid metal to cool to the temperature desired for handling of the units. Cooling to the point where the Hg solidifies is acceptable. However, the unit to be removed could be isolated by valves and with pressure applied, the liquid Mg can be expelled into a container. Temperatures acceptable by the container is first reached, then expulsion accomplished. With the malfunctioned unit replaced, charging of the system with the Hg will require heat application to maintain the liquid form. This will require a power supply for heaters, until the system is self-sufficient. The turbine can be returned to earth for rework.

Photovoltaic Cell

a. Failure Type

- (1) Deterioration or failure of solar cells due to extreme temperature decrease (prolonged eclipse); degraded cell performance due to radiation; sublimation of materials in the vacuum environment; degraded cell performance due to micrometeorites pitting and eroding surface.
- (2) Storage battery failures due to distortion of cells, structural failures, open welds causing leaks, electrode degradation, internal shorts and electrode corrosion and contamination.
- (3) Orientation system failures due to positioning motor bearing failure, deterioration and loss of lubricant, servo electronic failures.

b. Maintenance Requirements

- (1) Replace solar cell modules. Repair of modules can, if necessary, be accomplished within the pressurized space station. One or more cells can be replaced to restore the module to a serviceable condition, adhesive mounting of the cells appears to present no problem. Electrical connections to cell could be made by soldering with a soldering iron or by the use of a tube contained semiliquid solder. Use of a soldering iron would require caution to prevent solder from straying and adhering to exposed equipment in the maintenance room. Other electrical connections can be made using spot welding or wire wrap techniques. With wire wrap techniques, exposure to the vacuum can cause vacuum welding providing an additional electrical connection.

(2) Replace storage battery.

(3) Replace orientation system components, such as servo motors, amplifiers, relays, sun sensor, etc.

Chemical-Turboelectric

In view of the similarity between components of this power supply and many of those employed in a rocket propulsion system additional discussions are referred to the propulsion section of this report.

Fuel Cell

a. Failure Type

(1) Leakage of fluid line connections, gaskets; fuel cell structure failure; cell failure due to excess current drain; gas leakage; corrosion by electrolyte.

b. Maintenance Requirements

(1) Replace complete fuel cell module.

(2) Minor leakages might be corrected by retorquing

Dynamic Heat Engine

a. Failure Type

(1) Thermionic diode failure due to structural failures as a result of thermal stresses, crystallization of emitter material, cesium corrosion and emitter evaporation.

(2) Orientation system failures due to positioning motor bearing failure, deterioration and loss of lubricant, servo electronic failures.

b. Maintenance Requirements

(1) Replace thermionic converter module. These diodes are considered throw-away modules.

(2) Replace orientation system components such as servo motors, amplifiers, relays, sun sensor, etc.

Storage Batteries

a. Failure Type

- (1) Separator degradation, open welds causing leakage, distortion of cells, structural failures, electrode degradation, internal shorts, electrode corrosion and contamination, and leakage.

b. Maintenance Requirements

- (1) Replace storage battery. These units are considered as throw-away modules.

I. AVIONICS SYSTEMS

A significant amount of evidence has been presented in other sections of this report as well as in the literature regarding the importance of accomplishing rendezvous of two vehicles in a space orbit environment. In fact, the feasibility of ESWS, MTSS and the cislunar vehicle are directly dependent upon development of the concepts, techniques and equipment needed to perform this critical maneuver. It is immediately obvious then, that the majority of equipment provided to effect rendezvous will fall into the avionic category and should thus be analyzed for in-space maintenance requirements.

Communications and data transmission between orbiting vehicles attempting to rendezvous and from the earth to these same vehicles is also significant to the space programs. Here again, the maintenance requirements of equipment in the space-borne communications and data transmission network should be investigated for potential malfunction modes wherein restoration tasks become critical of accomplishment.

The ESWS and MTSS of this contract are permanent orbiting stations having inhabited life expectancies of one to five years. The prime literature sources on these two systems recommend incorporation of an artificial gravity state induced by rotating the vehicle about its principal axis. Unless this rotation and station stabilization is properly controlled by avionic elements, the entire operational structure could be effected.

In order to prevent decay of the orbital altitude of these large stations, it will be necessary to periodically add a velocity increment opposing this lowering. This increment must be accurately determined by onboard avionic equipment in terms of magnitude and direction before being introduced into the propulsion system.

The preceeding discussion serves to illustrate some of the more significant of the multitude of applications of avionics to the space vehicles of this contract. Since a prime objective was to investigate the maintenance requirements of various

space vehicle subsystems, the avionic components making up conceptual navigation and guidance, communications, attitude and stabilization and telemetering systems were analyzed in some detail. The following sections present significant aspects of this analysis.

1. Rendezvous Guidance

The schematic of Figure 19 has been constructed to illustrate the major functional aspects and equipment requirements necessary on a chaser vehicle such as Slomar which must effect a rendezvous with a target, say the MTSS. This network will function as follows: A beacon system onboard the target vehicle transmits a signal which is intercepted by the scanning beam of the rendezvous antenna of the chaser. Lock-on is effected and switch over to cooperative mode is accomplished. The chaser then interrogates the target beacon and derives range and range rate data. The major components of the receiver network consist of a mixer oscillator, IF amplifier and detector amplifier. The amplified signal is picked up by a decoder and fed to the range resolver which, in conjunction with gate generator pulse timing indications, produces a range and range rate data applicable for presentation on the display indicators in the crew compartment. The initial pulse generating network consisting of the trigger generator, pulse coder and transmitter modulator is also shown on Figure 19. Since a narrow beam rendezvous antenna is employed, it becomes possible, by interrogating the vehicle attitude reference, to measure antenna orientation with respect to the target and pick-off azimuth and elevation data relative to the two vehicles. In order to eliminate the need for a duplicate system, a set of broad beam antennae and associated transponders are provided on the chaser vehicle to permit earth tracking during orbital maneuvers.

Conversion and display of the data available from the rendezvous guidance system can be accomplished as shown in Figure 20. This schematic shows the capability for integrating into one unit display data from UV and IR scanners, Videcon, viewfinders and vehicle internal situation sensors.

2. Communications

The schematic shown on Figure 21 is an integrated system which incorporates voice transmission internally or between vehicles, as well as telemetry data transmission from the vehicle to the ground receiving station. Information from the telemetry sensors is sent to a signal conditioner, an encoder, the multiplexer controller and finally, transmitted through the diplexer and antenna. Voice transmission follows essentially the same path with the exception of passing into a voice encoder instead of the pulse multiplexer encoder. The vehicle internal communications network incorporates provisions for intercom functions as well as for transmitted and received audio presentation. Provisions for antenna orientation include a computer, mechanical pointing mechanisms and vehicle situation data pickups.

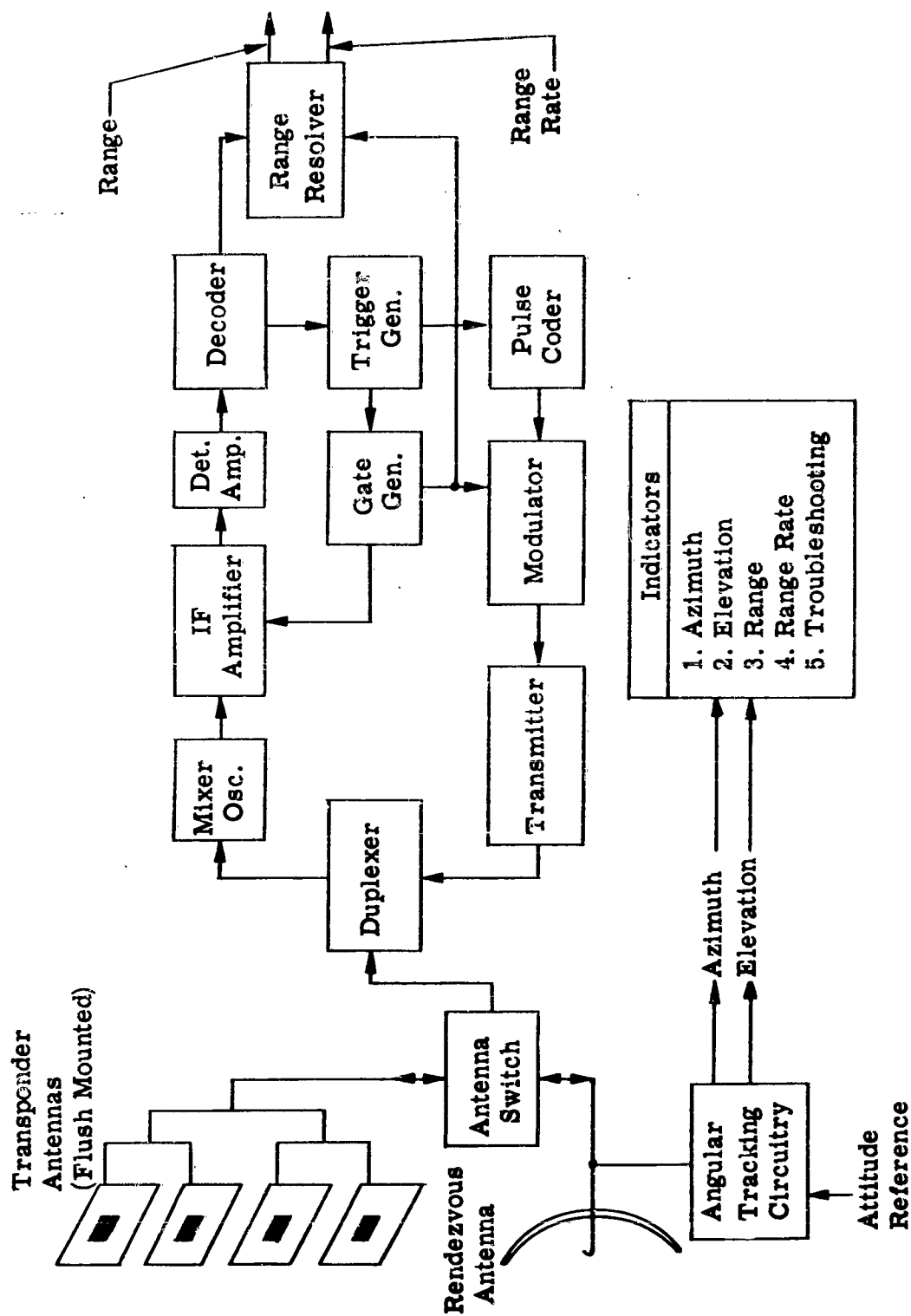


Figure 19. Rendezvous Guidance Functions

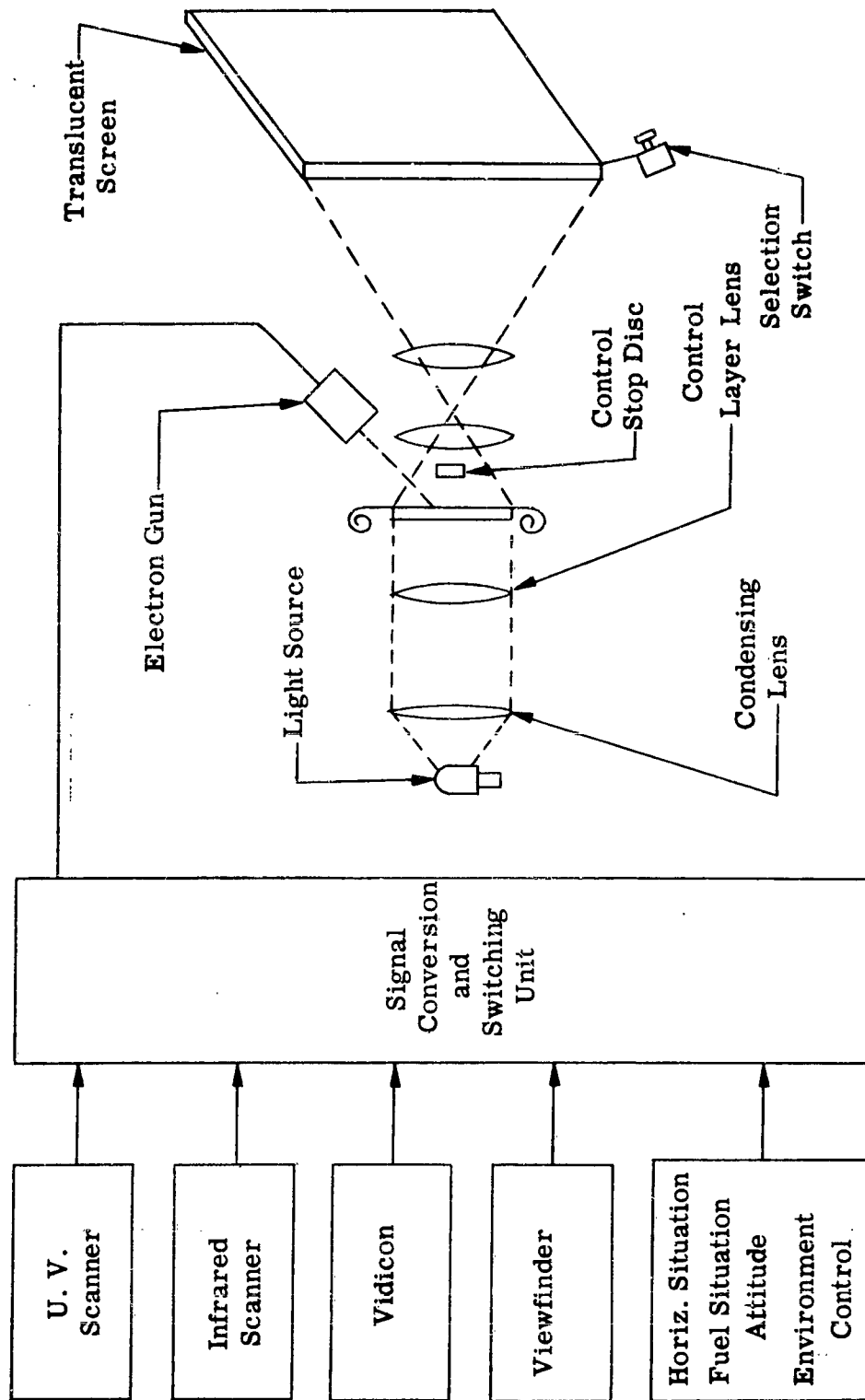
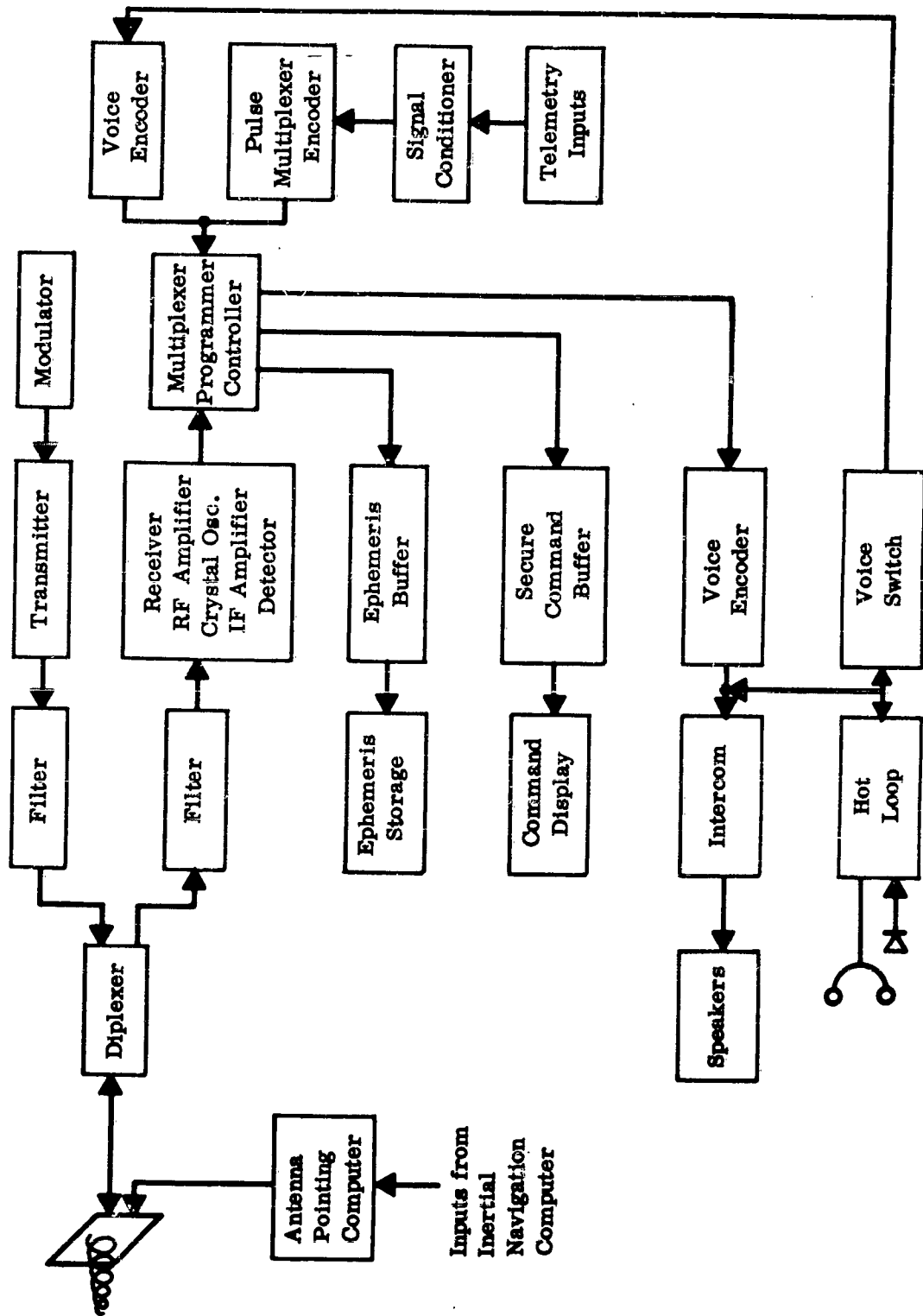


Figure 20. Integrated Display System



3. Attitude Control and Stabilization

The diagram in Figure 22 illustrates the major functional components required for the attitude control and stabilization functions of a space vehicle. The heart of this system is the computer which receives information from a data link between vehicles or between earth and station, a horizon scanner providing local verticals, inertial components, gyros, and a digital clock for the time reference. Interpretation of this data by the computer provides information for determination of flight path adjustments, attitude or spin rate changes as well as orbital position display data. Flight path changes and attitude or spin rate change as indicated by computer can be automatically accomplished by directing a command to the flight controller, and into the programmer which transmits appropriate thrust data to propulsion systems as required.

4. Maintenance Analysis - Avionics Systems

In the course of examining and analyzing maintenance tasks for the Avionic systems, it became apparent that the orbital worker may be called upon to repair many different types of equipments within a minimum time. Consequently, the worker would require a high degree of training in the theory and maintenance of electronic equipment as well as having at his disposal electronic schematics together with documented troubleshooting procedures. It is suggested that the overall task for the orbital worker could be reduced by including the use of a microfilm maintenance machine. This system would embody an adaptation of the existing VSMF Catalogue file manufactured for vendor equipment specifications wherein data is catalogued on microfilm in such a manner that a search for a particular item may be made in a matter of several minutes or less. The features of this machine which make it appropriate for space applications is that it can effect a tremendous savings in document weight, space, and search time. The complete machine consists of a display, film-reel driver, takeup spool, and microfilm cartridges. A search for information contained in the microfilm cartridges is made by inserting the correct cartridge into the machine, pressing a start button, and watching for the correct decimal coded stop mark to appear on the display. Once the correct index mark is found, further control can be effected to make the correct subindex mark together with a picture of the part or specifications appear on the display screen.

Adapting this machine for space vehicle repair and maintenance would consist of placing the complete space equipment schematics and troubleshooting procedures on microfilm cartridges. Specified equipment characteristics could then be compared with measured values and the proper repair be made. Repair time for any one vehicle subsystem could be minimized by combining present schematics into a more efficient nodal type. In this type of format, the points of common voltage and signal flow are grouped for easier initial location and functional servicing.

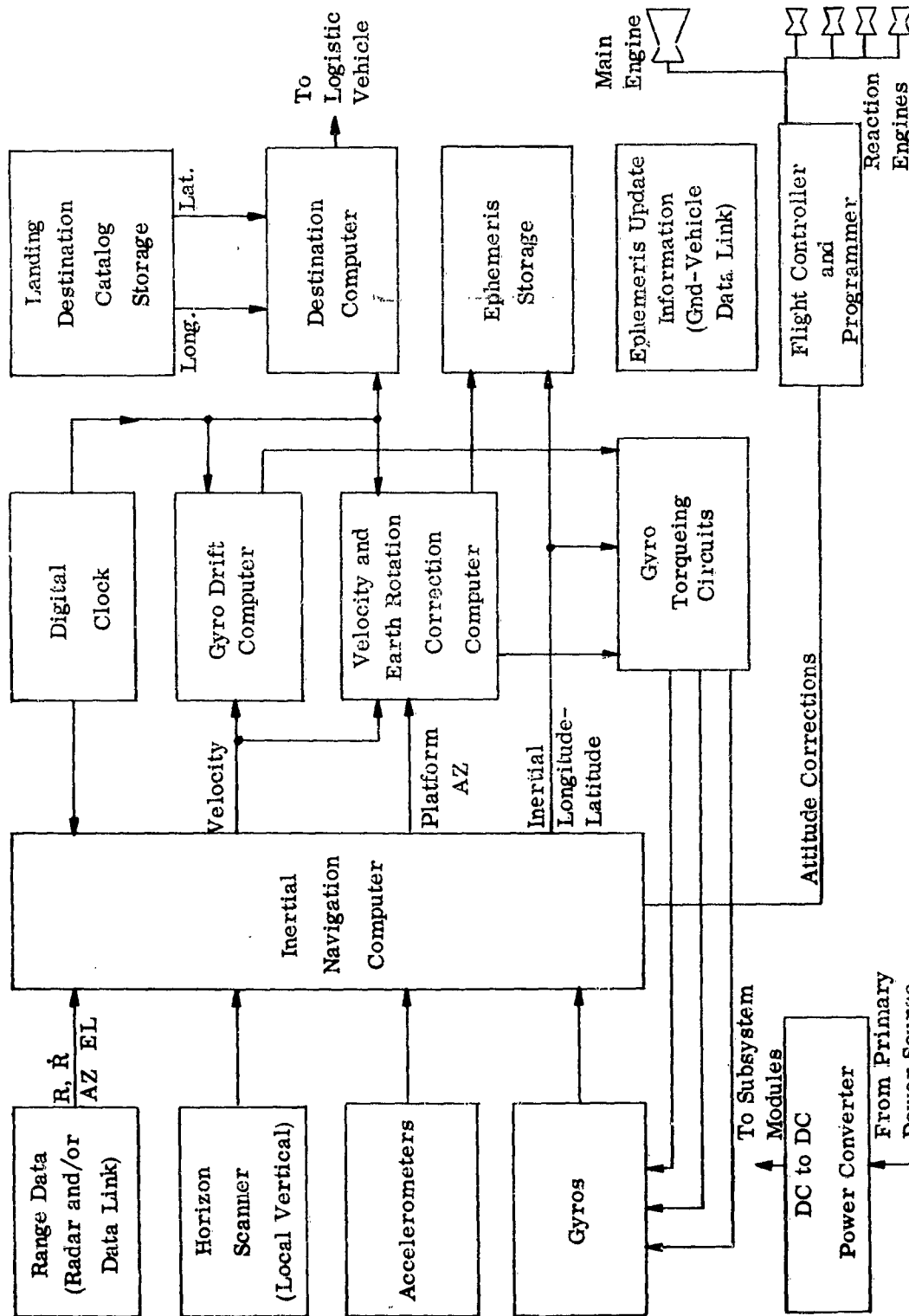


Figure 22. Attitude Control and Stabilization Functions

Another support system which would enhance the capabilities of the orbital worker is a universal test and maintenance console. This console would contain signal simulators, closed loop test sets, meters, and an all-purpose oscilloscope. By incorporating a few basic functional circuits and switches, the test console could supply test signals for most on-board electronic systems with a minimum of weight and power requirements. Use of printed circuits and micromineature components would further reduce the overall weight and power requirements.

The detailed maintenance analysis performed during the contract consisted of first identifying the major component or component group included in the principal avionic systems of the space vehicles, then examining these for failure modes, malfunction cues and verifications and the resulting maintenance requirements. It was discovered that the vast majority of component failures was caused by one or more of the following:

- (1) Loosening of component mount or material failure of mount and/or component. Surface distortion.
- (2) Shift in mechanical or optical alignment.
- (3) Physical jamming, rapid wear, erosion or brinelling of moving parts.
- (4) Environmental effects such as cold welding, development of high resistance joints, micrometeorite bombardment, change in solid state component parameters, outgassing, molecular breakdown, lubricant deterioration or change in insulation resistance.
- (5) Inadvertent operation of components caused by vibration, corrosion, oxidation or loosening of sub modules, cards or components.
- (6) Shorts or opens within a component, burnout of filaments or arcing within or around a component.
- (7) Increase in contact or output noise.
- (8) Excessive drift of component characteristics.
- (9) Change in component electrical value or submodule gain or signal bypass.
- (10) Saturation effects.

By classifying the failures which each identified component may experience, it was then feasible to establish the events which will occur as a result and the cues which can be employed for discriminating the malfunction. Once this has been accomplished, estimates were made of applicable verification techniques and the maintenance tasks required for system restoration and completion of the mission.

To illustrate the methodology employed and the depth of data generated in these areas during the contract, an abbreviated presentation is included in Table IV.

TABLE IV

<u>Subsystem</u>	<u>Subassembly</u>	<u>Malfunction Cues</u>	<u>Malfunction Verifications</u>	<u>Maintenance Requirement</u>
GUIDANCE FUNCTION				
Tracking Transponder	Radome	Visual Crack	Visual Inspection	Fill crack, replace radome
	Antenna (Flush)	High voltage circuit broken-opens	Perform antenna continuity check.	Dismantle and repair connector
	Duplexer	Power leakage into detector	Monitor power leakage on scope	Replace TR tube
	Detector	Erratic Beacon responses	Low pulse signal-to-noise ratio	Measure resistance replace diode
	IF Amplifier	Low gain	AGC changes, waveshape changes	Replace IF Amp. module or transistor
	Decoder	Erratic beacon triggering	Inject test signal, monitor waveshape	Replace delay line card
	Modulator	Overload light goes on	Perform continuity tests	Replace modulator tube
	Magnetron	Filament ind. light is off	Monitor filament current	Replace magnetron tube
	Waveguide	Intermittent plate current indication	Use videomonitor at waveguide test points	Replace waveguide seal
	Power Converter	Overload limit light goes on	Test point continuity checks	Repair terminal board - Replace component
Rendezvous Radar	Dish Antenna	Servo Drive overload light goes on.	Monitor dials and inspect visually	Weld antenna mount
	Antenna Switch	Low output power	Monitor with test video detector	Replace contacts or complete switch
	Diode Mixer	Erratic beacon triggering	Monitor injected test signal	Replace crystal diode
	Modulator	Modulator current drops to zero.	Continuity check of coil	Replace coil
	Magnetron	Circuit breaker opens	Monitor waveform after WFO on	Replace magnetron
	Waveguide	Intermittent plate current	Use video monitor at waveguide test points	Replace waveguide or seal
	Range Synchros	Delayed range lock on of test signal.	Monitor injected test signal	Replace synchro
	Indicators	-	Monitor injected test parameter	Replace meter or tube
	Servo Amplifiers	Erratic servo amp outputs	Monitor performance of test signal	Repair or replace amp submodules
	Power Converter	Overload limit light goes on	Test point continuity checks	Repair terminal board replace component
Stabilized Platform	Gate Generator	Distorted waveform on monitor	Monitor test points on submodules	Repair or replace amp submodules
	Gyros	Drift from star reference	Measure drift angle-stars or stored ephemer.	Replace gyro-activate and align spare platform
	Gyro Motors	Vibration transducer output	Inspection (Stethoscope)	Replace gyro motor
	Torqueing Computer	Unusual torque indications	Monitor results of injected test signal.	Replace computer module
	Torqueing Amplifier	Unusual torque indications	Monitor results of injected test signal.	Replace computer module
	Torqueing Motors	-	Monitor results of injected test signal.	Repair or replace.
Stabilized Platform	Digital Clock	Negative Parity Check indication	Check against duplicate clock or earth transmitted signal.	Repair transistor diode or replace module
	Inertial Computer	Checkout test program erroneous	Troubleshoot computer sub-routines	Replace submodules or cards.
	Operational AMPs	Low output signal on monitor	Monitor standard test signal	Replace capacitor
	Power Converter	Overload limit light goes on	Test point continuity checks	Repair terminal board-replace component

CHAPTER 4

MAINTENANCE TASK ANALYSES

Although the Maintenance Analyses described in the preceding sections of the report have defined potential malfunctions and thus the specific requirements for performing Maintenance, it still remains to determine how the maintenance can be accomplished. This determination is dependent upon the delineation of specific activities that must be accomplished in the fulfillment of the maintenance requirements that have been identified. To this end, a maintenance task analysis was undertaken to describe the sequence of events that must transpire in the restoration of malfunctioning equipments to a normal operating state.

Even though the maintenance analysis was limited to 7 subsystem areas, the maintenance requirements that were identified were too numerous to permit an analysis of each. Thus an attempt was made to identify and rank order the sources of malfunction on the basis of expected occurrence and to limit the task analyses to tasks that are within the predominate area. Table V presents a rank ordering of the major malfunction sources for each of the subsystem areas.

TABLE V

MALFUNCTION SOURCES

Malfunction Source	Subsystem Area					
	Rendezvous Docking and Assembly	Propulsion	Avionics	Structure	Power Supplies	Life Support
Human Error	1	2	1	2	2	2
Component (1) Failure	2	5	2		4	3
Material Defect	3	6	3	4	5	5
Leakage		1			1	1
Contamination		3				6
Environmental		4		5	3	4
Meteorite Damage				1		
Structural				3		

(1) includes component failure resulting from shock, temperature, particle radiation, vacuum, vibration.

In selecting specific tasks for further analysis an attempt was made to select tasks that fell into the higher ranking sources and, at the same time, sample both internal and external maintenance activities. Table VI is a listing of the tasks that were selected for detailed analysis.

TABLE VI
MAINTENANCE TASKS SELECTED FOR DETAILED ANALYSIS

Subsystem Area	Selected Task
Rendezvous and Docking	Replace damaged switch in docking body of MTSS (extravehicular task)
Space Station Assembly	Assemble modules of MTSS with manned shuttle vehicles (intravehicular task)
Propulsion	Correct leakage in Agena gas generator injector (extravehicular task)
Avionics	Remove and replace a defective magnetron in the ESWS tracking transponder (extravehicular task)
Structure	Repair a meteorite puncture in the ESWS (extravehicular task)
Power Supplies	Remove and replace a fuel cell of the Lunar Landing Vehicle (intravehicular task)
Life Support System	Correct leakage in the plumbing network (intravehicular task)

It will be noted from this listing that it includes a task from the source considered most likely to occur in 3 of the 6 cases (ignoring assembly for the moment). In two other cases it was felt most advantageous to use the second most likely occurrence since the predominant source was judged to be human error which, at the present time, appears too nebulous to attempt to analyze in detail. In the remain case (power supplies) a low ranked area was utilized. This was the result of the following considerations: (1) leakage correction is essentially the same as the other leakage problems; (2) an internal task was desired; and (3) human error was bypassed for the reason given above.

In the conduct of the task analysis, major consideration was given to each of the following variables:

- (1) The maintenance requirement as derived from the overall maintenance analysis.
- (2) Accessibility of the part to be repaired.

- (3) Detailed task activities required for removal, repair and/or replacement.
- (4) Task times
- (5) Number of kinds of fasteners
- (6) Tool requirements
- (7) Forces required for manipulation of tools and parts.

The results of these analyses are presented in Tables VII through XI. The time estimates contained in the tables are based on performance of selected task items on a frictionless floor and on available data from the literature. As is pointed out later in the report, they do not include degradations in performance expected from a pressure suit or other methods of encapsulation.

Examination of these data does not reveal any tasks that appear to present unsurmountable difficulties. Although, unless some greater standardization of methods of fastening is achieved, performing maintenance will require a considerable number of tools. Even for the limited number of tasks analyzed in this study, the following list of tools was identified as required.

<u>Quantity</u>	<u>Tool</u>
1	1/4 inch socket wrench
1	3/16 inch socket wrench
1	3/16 inch Allen wrench
2	3/8 inch open end wrenches
1	1/2 inch open end wrench
2	5/8 inch end wrenches
1	1 inch end wrench
1	Phillips head screwdriver
1	Slot head screwdriver
1	Long nose pliers
1	Lock wire cutting tool
1	Camlock tool
1	Multimeter
1	Drill
1	Hole saw
1	Rivet gun
1	Small welder
1	Electronic test set
1	Flow meter

Thus the accomplishment of these six tasks would require approximately 21 separate pieces of tools and equipment. Another consideration is the number of pieces that must be handled by the worker in performing work outside the vehicle. Including tools these range, for the external tasks, from a minimum of 10 to a

maximum of 38 for the tasks analyzed. To assure availability of these items when needed, some means will be required for securing them in this work area.

Finally, as will be discussed in the Human Capabilities, Tools and Fasteners, and Remote Manipulator Chapters, the torquing requirements do not appear to be beyond the capability of the worker if proper tie-down points and/or hand holds are available.

TABLE VII

MAINTENANCE TASK ANALYSIS

Task: Replace damaged switch in docking boom of MTSS (Rendezvous-Docking). This task does not appear to present any problems of accessibility since the switch lies in the bottom of an open disk.					
Task Requirements	Estimated Time (Accumulative) Hrs/Min.	Standard Fasteners		Force Data Torque In. Lb	Tool Requirement
		Qty.	Ident.		
1. Don soft suit equipment	00:00				
2. Check operation of life support, communications and in "Tied Down" condition check maneuvering system.	00:30 00:40				
3. Secure tools and spare parts	00:41				
4. Enter air lock					
5. Secure tether cable	00:42				
6. Evacuate air lock					
7. Check operation of life support equipment and communications					
8. Check suit leakage	00:52				
9. Open air lock outer door					
10. Maneuver to conical end of boom	00:54				
11. Attach mooring lines	00:55				
12. Insert shorting plug (thrusters)	00:57				None

TABLE VII (CONT)

Task Requirements	Estimated Time (Accumulative) Hrs/Min.	Standard Fasteners		Force Data Torque In. Lb	Tool Requirement
		Qty.	Ident.		
13. Remove access plate fasteners and stow	00:57	4	AN509-8-32	12-15	Phillips Head Screw Driver
14. Withdraw switch assembly	00:58				None
15. Disconnect MS connector					AN/PSM-6 Multi-meter
16. Continuity check switch (operate manually) and measure contact resistance	01:02				Two end wrenches
17. Remove switch mounting fasteners and stow	01:03	2	5/8 NUT		
18. Remove switch and stow					
19. Install replacement switch					
20. Install switch mounting fasteners	01:06	2	5/8 NUT		Same as 17
21. Connect MS connector	01:07				None
22. Install switch assembly					
23. Install access plate fasteners	01:10	4	AN509-8-32	12-15	Same as 13
24. Stow tools and test equipment					None
25. Remove shorting plug and stow					
26. Detach mooring lines	01:12				
27. Return to air lock	01:13				

TABLE VII (CONT)

Task Requirements	Estimated Time (Accumulative) Hrs/Min.	Standard Fasteners		Force Data Torque In. Lb	Tool Requirement
		Qty.	Ident.		
28. Secure air lock outer door	01:14				
29. Pressurize air lock					
30. Remove tether cable	01:17				
31. Exit air lock	01:18				
32. Remove tools, replaced part(s) and test equipment					
33. Remove soft suit equipment	01:33				

TABLE VIII
SPACE STATION ASSEMBLY TASK ANALYSIS

Task: Assembly of a modular space station using one man shuttle vehicles*.

1. Each of the two shuttles proceed to retrieve a module and position it at the assembly point.
 - a. Module is located and identified.
 - b. Shuttle translates to module.
 - c. Shuttle secures its attachment arms to standard fittings mounted on end of module opposite each end designated for assembly.
 - d. Shuttle positioned so that the shuttle's thrust is through the module's center of mass.

To enable proper positioning, standard fittings for shuttle attachment are mounted at proper locations at each end of a cylindrical module and at the side. The center of mass at each location is marked. (Modules of reference multimanned space station have external rails mounted for manipulator attachment.)
 - e. Shuttle pushes module to assembly point and positions assembly end in close proximity (approximately 20 feet) to mating module and at a zero relative velocity.
2. Shuttle, still attached to retrieved module, applies proper roll thrust to module for gross alignment with mating module. Second shuttle stationed at the mating module provides indexing information.
3. Closing force (push) is applied to position module so that second shuttle can grasp both modules with its attachment arms. This attachment is made such that the second shuttle can act as a pivot.
4. Shuttle 1 applies appropriate thrust to the retrieved module to longitudinally align it with the mating module.
5. Shuttle 1 releases hold of retrieved module and joins shuttle 2, grasping both modules.
6. Both shuttles index modules to finer degree.

*Because of the undefined orbital maneuver and propulsion system capability of the shuttle vehicle, no attempt was made to specify task times.

TABLE VIII (CONT.)

7. Both shuttles apply closing force to modules, drawing them together, until contact is made and temporary latches around the periphery of the module have all engaged.
8. Allow modules to remain connected with temporary latching until the differences in temperature of the mating sections adjust.
9. With both shuttles again grasping both modules on opposite sides of joint, apply closure force, making sure that alignment pins enter conical lead-in mating hole, and press the two modules together.
10. With closure force still applied, slip the two halves of the peripheral clamping ring over the mated flanges.
11. Install tension bolts and tighten. Bolts are 1/2" diameter and require 480-960 in. lb of torque. A ratchet type socket wrench will be required.

NOTE: A seal which is integral with the end flanges is placed so that when the volume between the cylinders is pressurized, the air forces the seal tightly into the notch provided for it under the flanges, sealing the joint. Later, a more permanent seal is added from inside the pressurized station.
12. Proceed to next pair of modules and repeat assembly sequence.

NOTE: After all the modules are connected, proceed with shuttles to airlocks, space suited crew enter space station and begin installation procedure.
13. Pressurize volume between each module (open valve allowing air from pressurized module compartments to flow into volume between modules, thereby pressurizing them).
14. Install module interconnecting electrical cabling (MS connectors, Class E).
15. Install module interconnecting hydraulic lines (assuming one inch flex lines requiring 70-1150 in. lb. of tightening torque**).
16. Install module interconnecting air ducts (nylon reinforced plastic tubing, clamped to module ducts, tape sealed).

Subsequently, power is turned on, atmosphere regeneration system is placed into operation and station equipment is placed into operating position and condition.

**Torque data obtained from Bell Process Specification BPS 4018, Rev. M. 10-6-58; Amendment 1, 2-9-61.

TABLE IX
MAINTENANCE TASK ANALYSIS (PROPULSION)

TASK: Replace leaking gas generator injector gasket in turbine pump (See item 4). Access to two bolts will present problems since they are under flexhoses and will require a reach of approximately 12 inches. The optimum location for accomplishing this torquing task would be straddle of the engine thrust chamber facing toward the gas generator. Locking the feet around the thrust chamber would help provide a reaction for the forces developed during the maintenance operation.					
Task Requirements	Elapsed Time Per Task	Standard Fasteners	Force Data	Tool Requirements	Other Data
1. Don suit, check out suit equipment, secure tools and spares, enter air lock, secure tether, egress air lock, check suit operation and maneuver to the malfunction.	54 min.	1. None (except on suit)	1. None (except for those required while donning suit and egressing from air lock)	1. None	
2. Secure solid charge igniters by cutting lockwire and separating electrical connectors. There are two igniters per start can, 4 total connectors are fitted 7-505-9.	16 min.	2. Lockwire and electrical connectors.	2. Closed force squeeze action 1 pound approximately. Force	2. Lockwire cutting tool. Connectors will be removed manually by torquing the screw collars.	2. The connector wrench designed for use in space would be useful for this task.
3. Remove two 3/16" bolts from start can assembly clamps, stow bolts.	18 min.	3. AN 3-11 Bolts (2)	3. 20-25 pounds (Torques were obtained from BPS 4018)	3. Open, box, or socket wrench.	3. A closed force torqueless tool is indicated.
4. Remove two 3/16" bolts from start can assembly Bracket, stow bolts	0.8 min.	4. 6Z1-1032C10 bolts (2)	4. Same as 3	4. Allen Head wrench.	4. Same as 3
5. Disconnect two flex hoses from gas generator injector.	1.3 min.	5. AN818-6C nut 3/8"	5. 100-250 pounds	5. Two open end wrenches, one for holding and one for torquing.	5. Access to these tube nuts is difficult. A special tool is also indicated due to the need for coordinated motions in nut removal.
6. Remove ten (10) 6Z1-1032C-10 bolts from top of gas generator injector, stow bolts.	8.0 min.	6. 6Z1-1032C-10 bolts (10)	6. 20-25 pounds for removal 35-40 pounds for installation.	6. Allen Head wrench.	6. --
7. Remove complete gas generator starter assembly (stow the assembly)	0.5 min.	7. None	7. Some force may be required to separate assemblies from each other.	7. A screw driver or a prying tool might assist the separation task.	
8. Remove gasket	0.5 min.	8. None	8. None	8. A screw driver or thin blade tool is needed to remove gasket from groove.	
9. Install new gasket by following reverse task order 7-2.	12.0 min.	9. See 7-2	9. See 7-2	9. See 7-2	9. The (10) 6Z1-1032C bolts are first torqued up to 35-40 pounds, then backed off and retorqued to 20-25 pounds.
10. Return to airlock enter, pressurize, doff suit, etc.	20.0 min				

TABLE X
BLACK BOX ASSEMBLY - ESW TRACKING TRANSPONDER

TASK: Replace Magnetron Tube. Although this package would normally be mounted near the radiating antennas, it is anticipated that an access panel would have to be removed. In order to accomplish the torquing task, the operator may require a closed-force torqueless tool or a technique for producing a counteraction of his body and the vehicle skin. (Possible use of mounted handles) Note: Tasks 2 through 6 would be performed in an external environment whereas Tasks 6 through 17 would be performed in an internal or shirt-sleeves environment.					
Task Requirements	Elapsed Time/Task	Standard Fasteners	Force Data	Tool Requirements	Other Data
1. Don suit, check out suit equipment, secure tools and spares, enter air lock secure tether, egress air lock, check suit operation and maneuver to the malfunction.	94 min.	1. None (except on suit)	1 --	1. None	1. None
2. Remove access panel by loosening fasteners similar to camlock type. (Total number - approximately nine). Slow panel.	6.0	2. Camlock	2. 10 lb in.	2. Camlock tool	2. Camlock screw heads should be adaptable for use by suited astronaut or remote manipulator.
3. Unscrew electrical power connector.	1.0	3. AN screw type	3. 3 lb in.	3. Lockwire cutting tool. Connectors will be removed manually, by torquing the connector collar.	3. A connector wrench designed for use in space would be useful for this task.
4. Unscrew RF connector (one)	1.0	4. Type (N) connectors	4. 2 lb in.	4. Connector loosened by use of open end wrench.	4. Possible use of a quick disconnect connector would preclude the need for tools
5. Remove transponder case from heat sink mounting by removing six nuts from assembly flange. Pull can and flange off mounting studs, place nuts in suit receptacle.	4.0	5. 3/8 inch nuts	5. 75 lb in.	5. Two open end wrenches, one for holding and one for torquing.	5. None
6. Transport and maneuver the transponder package to repair section of vehicle or orbiting repair facility. Remove suit - commence repair task.	30.0	6. Carrier clamp on suit or compartment within a shuttle capsule.	6. None	6. Special purpose clamp or quick release friction strap.	6. Friction belt similar to an auto belt may be adapted for this purpose.
7. Remove transponder chassis out of case by first extracting six Phillips head machine screws.	6.0	7. Phillips machine screws	7. 15 lb in.	7. Phillips screwdriver	7. Snap lever clamp might be used to an advantage here in order to save repair time.
8. Loosen R.F. connector collar on magnetron tube	1.0	8. Type (N) RF connector	8. 2 lb in.	8. Same as No. 4	8. None.
9. Remove four mounting screws on base of tube.	1.0	9. None	9. 0.5 lb in.	9. Same as No. 7	9. None
10. Cut filament and high voltage leads at the terminal board mounting lugs, remove defective tube.	2.0	10. None	10. --	10. Long nose wire cutting tool	10. Use of pre-welded lugs at this point in the task could simplify subsequent installation of tube.
11. Insert new magnetron into mounting position. Note: (Measure filament continuity and for plate lead shorts before installing.)	1.0	11. None	11. Manual operation	11. None	11. Possible use of tinker toy plug-in concept on the tube pins would alleviate the removal and insertion.
12. Tighten type N collar on magnetron tube output		12. Same as No. 8	12. Same as No. 8	12. Same as No. 8	12. None
13. Connect tube leads and weld		13. None	13. Manual operation.	13. Small table top resistance welder.	13. Small retainer clamp may be necessary to hold wire leads in position during welding.
14. Insert four mounting screws on tube mount.	1.0	14. None	14. Manual operation.	14. Phillips screwdriver	14. None
15. Insert transponder chassis into mounting case, replace and tighten screws.	2.5	15. Phillips machine screws	15. Same as No. 7	15. Same as No. 7	15. None
16. Perform bench check: a) Connect power leads b) Connect dummy load c) Connect power meter d) Connect interrogator e) Locate malfunction f) Correct malfunction	30.0	16. a) power cable with connector b) RF cable with connectors c) Bench mounted chassis clamps	16. Manual operation	16. a) Standard screwdriver b) Dummy loads c) Power meter d) C-band signal generator e) Pulse encoder f) Directional coupler g) Isolator	16. Use of small compact test set would eliminate bulky test equipment.
17. Repeat task No. 1 Repeat steps No. 6 through 8 with the exception that the tanks are assembly in place of removal. Then return to station.	66.0	17. None	17. Manual operation.	17. Space suit equipment	17. None

TABLE XI
COMPONENT - OUTER WALL OF ESWs MANNED CAPSULE

TASK: Repair Meteorite Puncture in Outer Wall of ESWs Manned Capsule. Accessibility to most areas of the manned capsule will be good.					
Task Requirements	Elapsed Time Per Task	Standard Fasteners	Force Data	Tool Requirements	Other Data
1. Don suit, check out suit equipment, secure tools, enter air lock, secure tether, egress air lock, check suit operation and maneuver to the area requiring repair.	54 min.	Suit Fasteners	1. Those required while donning suit and other tasks preliminary to the actual repair task.	1. None	1. --
2. Cut away outer wall and insulation.	8 min.	---	2. Several pounds of undirectional force will be needed.	2. Power driven hole saw, router, or rotary file.	2. A standardized hole template may be required for precise hole repairs.
3. Install prefabricated hole plug.	1 min.	---	3. Small	3. None	3. ---
4. Drill holes for plug fasteners to match predrilled plug.	3 min.	---	4. 2 pounds axial	4. Power drill	4. The use of a torque producing tool may be difficult even if the worker is restrained. New designs are indicated.
5. Apply sealing compound, insert plug and install fasteners.	12 min.	5. Rivets	5. Using a blind rivet tool a squeeze force less than 1 pound is indicated.	5. Blind rivet gun	5. Use of adhesives or welds instead of mechanical fasteners may be advantageous for repairs of this type. See discussion of possible fastening methods.
6. Return to airlock, enter pressurize, don suit, secure tools, etc.					

TABLE XII

MAINTENANCE TASK ANALYSIS

Task: Fuel cell replacement. Assuming an ion exchange membrane fuel cell the cell modules are separate assemblies. The maintenance as performed within a pressurized vehicle.					
Task Requirements	Estimated Time (Accumulative) Hrs/Min.	Standard Fasteners		Force Data Torque In. Lb.	Tool Requirement
		Qty.	Ident.		
1. Shut-off pressure source valve, open pressure, pressure vent valve, and shut-off hydrogen and oxygen supply valves.	00:00 00:02	4		It is considered that the cell is not be repaired at the station. It can be returned to earth for rework or disposed of a throw-away module.	
2. Check water, remove valve is closed, open cell vent valve, and shut-off hydrogen and oxygen supply valves.	00:07				
3. Disconnect electrical connector, MS type.	00:09				
4. Disconnect fuel line and oxygen line.	00:15				
5. Disconnect water remove lines.	00:20				
6. Remove fuel cell assembly mounting bolts.	00:32				
7. Remove fuel cell assembly.	00:37				
8. Install replacement fuel cell assembly, and mounting bolts.	00:42				
9. Connect water remove lines.	00:47				
10. Connect fuel line and oxygen line.	00:55				
11. Connect electrical connector	00:57				
12. Close pressure vent valve, open pressure source valve, open hydrogen and oxygen supply valves.	01:00				
13. Monitor flow rate and electrical output.	01:05				
14. Check system for leakages.	01:10				
		1 inch fitting	End wrench		
		1/2 inch fitting	End wrench		
		AN 520-1/4-28	Socket wrench		
		AN 520-1/4-28	Socket wrench		
			Same as 5		
			Same as 4		
			Flow meter VTVM (part of assembly.)		

CHAPTER 5

EFFECTS OF SPACE ENVIRONMENTS ON SPACE MAINTENANCE

A. GENERAL

This section discusses space environments that will influence the task of maintenance techniques in space. A wealth of data on such environments has been compiled within the last few years, based on evaluation of satellite, rocket, and balloon experiments. Many investigations on the effects of individual or combined space environments have also been performed under a variety of simulated conditions. An evaluation of the results show that five environmental parameters should be considered in the design and maintenance of space systems:

- (1) Zero gravity
- (2) Electromagnetic Radiation
- (3) Particle Radiation
- (4) High Vacuum
- (5) Meteoroids

The interaction of these parameters with the structure, subsystems, and components, and also the resulting biological hazards, are, in some cases, critical. Conversely, there are interactions of minor importance that will not cause severe malfunction and, in addition, are well known. Consequently, these interactions can be designed for in the early design stages and do not present critical problems to space maintenance techniques.

A typical example is the influence of the ultraviolet and X-ray spectral region of electromagnetic radiation primarily originating from the sun. We know from laboratory experiments that metals and alloys are not affected by solar illumination in space, and that the influences on the electrical and mechanical properties of organics is negligible. Semi-conductors, sensitive to particle radiation, also remain unaffected. Sunlight does effect the optical properties of insulators, and this must be considered for optical systems or windows where a constant transmission is desired. However, a transmission reduction will not limit any vital function and might also be reduced by proper design. The effect of the visual portion of the electromagnetic spectrum is discussed in the Human Capabilities Chapter.

Similar considerations apply to effects attributed to low or zero gravity. The behavior of liquid propellants, for example, at conditions of near zero gravity have been studied in simulation experiments, and proper means for eliminating undesired expulsion behavior can be provided during the design. Consequently, it was decided not to investigate low or zero gravity in relation to space maintenance requirements.

Its relation to the human capability to perform maintenance tasks in space is discussed in the Human Capabilities chapter of this report.

The environmental studies were restricted, for the above stated reasons, to three critical environments:

- (1) Micrometeorite impact
- (2) Particle radiation
- (3) Vacuum

Two areas have been considered; the effects of these parameters on the malfunction probability or performance degradation of critical system components and the biological hazards to the space maintenance worker as he attempts to engage in the required tasks.

The degree of criticality of the three selected environments depends on the mission (inclination, altitude, duration) but also on the configuration of the system. For example, the integrated external dose from particle radiation can be calculated from the mission parameters of the system. To determine the internal dose rate or integrated dose (both quantities are significant to radiation sensitive components or the astronaut), the structure of the system must be taken into account. The same considerations apply to micrometeoroids. Insofar as possible, the structural designs of the space systems studied during the contract were used as the basis of the analysis of the effects of the environments.

The first objective of the following environmental study was a qualitative and quantitative description of the characteristic phenomena of each critical environment, based on published reports and also on results made available for this study by NASA. In particular, the latest data (not published) on the mass distribution of micrometeoroids and the results obtained by Explorer XII on the spatial distribution of electrons of the outer belt to a distance of 80,000 km were included in the evaluation. In the second phase, the resulting effects were considered. Again, all results available from the literature were used, supported by a number of analytical studies in cases where an extension appeared to be necessary. Special attention was given to the calculation of internal doses due to particle radiation and to the different phenomena associated with micrometeoroid impacts (cratering penetration and spalling, and pressure loss in a pressurized cabin).

B. METEOROIDS AND THEIR EFFECTS ON SPACE STRUCTURES

Encounters with space debris will directly affect the mission life of the space systems which must be maintained, the maintenance tasks required of the orbital workers and the design characteristics of the extravehicular protective systems employed by the maintenance worker. Three direct effects of micrometeorite impact are apparent:

- (1) Sandblasting walls and windows by the continuous impact flux which will occur.

- (2) Damage to the structural integrity of the vehicle and damage to internal vehicle systems or occupants due to spalling (breaking-off) of the walls when impacted by a particle with insufficient momentum to penetrate.
- (3) Structural damage and damage to internal vehicle systems and occupants due to penetration by a meteorite.

The sandblasting effect of meteorites might require replacing damaged windows at certain intervals for vehicles with long mission durations. Surface erosion of structural walls, however, does not appear to be a significant factor in system life and, therefore, in orbital maintenance requirements. Spalling and penetration, on the other hand, are potentially serious problems relative to space maintenance and are covered in detail in the following pages.

1. Particle Characteristics

Estimates of meteoroid particle densities range from 0.05 g/cm^3 , which implies a lacy, porous structure, to that of iron, which is 7.0 g/cm^3 . It appears to be a reasonable and conservative assumption that most meteoroid particles will consist of stoney material⁽³⁰⁾. The density of such particles is approximately that of aluminum, or about 2.7 g/cm^3 .

Impact velocities are generally believed to range from 11 km/sec to 72 km/sec in the vicinity of the earth. At the present time, little is known in regard to particle velocity vectors⁽³¹⁾ and the distribution of meteor velocity. A mean value of 40 km/sec is used by some authors as a nominally high value, and is assumed, in the analyses presented in this report, for the purpose of a conservative estimate of meteoroid effects. A probability distribution model which includes the stochastic variation of particle velocity is necessary to reach optimum accuracy in predictions of micrometeoroid penetration. Such a model must await more up-to-date information on the velocity distribution before it can be validly developed, however.

With given density and mass of a particle, one may calculate its diameter under the assumption that the particle is spherical. Hence, all the relationships given below in terms of particle mass may also be presented in terms of particle diameter. In particular, the penetration equation, given below as a function of mass and velocity, may similarly be presented as a function of particle diameter. A velocity of 40 km/sec and the density of aluminum are assumed.

2. Meteor Flux Density and Mass

The spatial and temporal distribution of meteorite velocities (in any system of reference) is not known, except for some regularly occurring streams. At the time such streams occur, the collision risk increases by several orders of magnitude. It would be desirable to confine space vehicle operations to relatively inactive stream periods near the earth. For such periods, the meteoroid encounters are randomly distributed.

Recent information on flux density near earth has been acquired from Goddard Space Flight Center⁽³¹⁾, in connection with this contract. From these data, which reflect the results of various American and Soviet rocket experiments, the relationship between flux density F and meteoroid mass may be stated as:

$$F = 10^{-12} M^{-1.1} / \text{m}^2\text{-sec} \quad (M > 10^{-8.5} \text{ grams}) \quad (1)$$

and

$$F = 10^{-17} M^{-1.70} / \text{m}^2\text{-sec} \quad (10^{-10} \text{ grams} \leq M \leq 10^{-8.5} \text{ grams}) \quad (2)$$

F is defined as the flux density (in number of particles per square meter and second), having a mass greater than m grams. This relationship is presented in Figure 23. The following calculations for space vehicle wall thickness and penetration probability are based on this mass distribution law. The results are considered to be conservative.

3. Impact Probability

Equations (1) and (2) may be used to calculate the total number of micro-meteoroid impacts on a vehicle during its mission. Figure 24 illustrates the results of such calculations assuming mission times of 1 day and 5 years and vehicle surface exposed areas of 2.5 to 300 square meters. It should be noted that, in this figure, the total number of impacts refers to particles with a mass greater than the specified value of the abscissa. Figure 25 presents the number of impacts for exposed areas and mission times representative of orbital space systems. The large number of impacts predicted for the 4-hour extravehicular suit and space maintenance capsule are especially interesting. Approximately 250 impacts of particles of 1×10^{-9} gm or greater are predicted for a 2.5 m^2 extravehicular suit and 1000 impacts of similar mass particles are predicted for the 10 m^2 space maintenance capsule. The implication of these high impact fluxes on the employment of these systems for space maintenance activities is developed in later sections of this report.

4. Impact Phenomena

The kinetic energy of a micrometeorite will be changed after its encounter with the vehicle. A portion of the energy will be dissipated by impact shock waves created within the wall and the micrometeorite which will result in heating the transmitting medium. Reflection of the wall shock at the inner surface will result in a local concentration of stresses that may be sufficient to cause particles to break off the inner wall (called spalling), even though the micrometeorite itself does not penetrate. Similarly, such a stress concentration within the micrometeorite would tend to pulverize it. Energy would also be consumed in the irreversible deformation of the wall and of the micrometeorite itself, as well as overcoming the frictional effects of the latter's passage through the wall. An encounter with a sufficiently energetic micrometeorite would result in, in addition to spalling, a penetration of the wall. The kinetic energy of the micrometeorite after penetration would be the energy of the oncoming particle less the sum

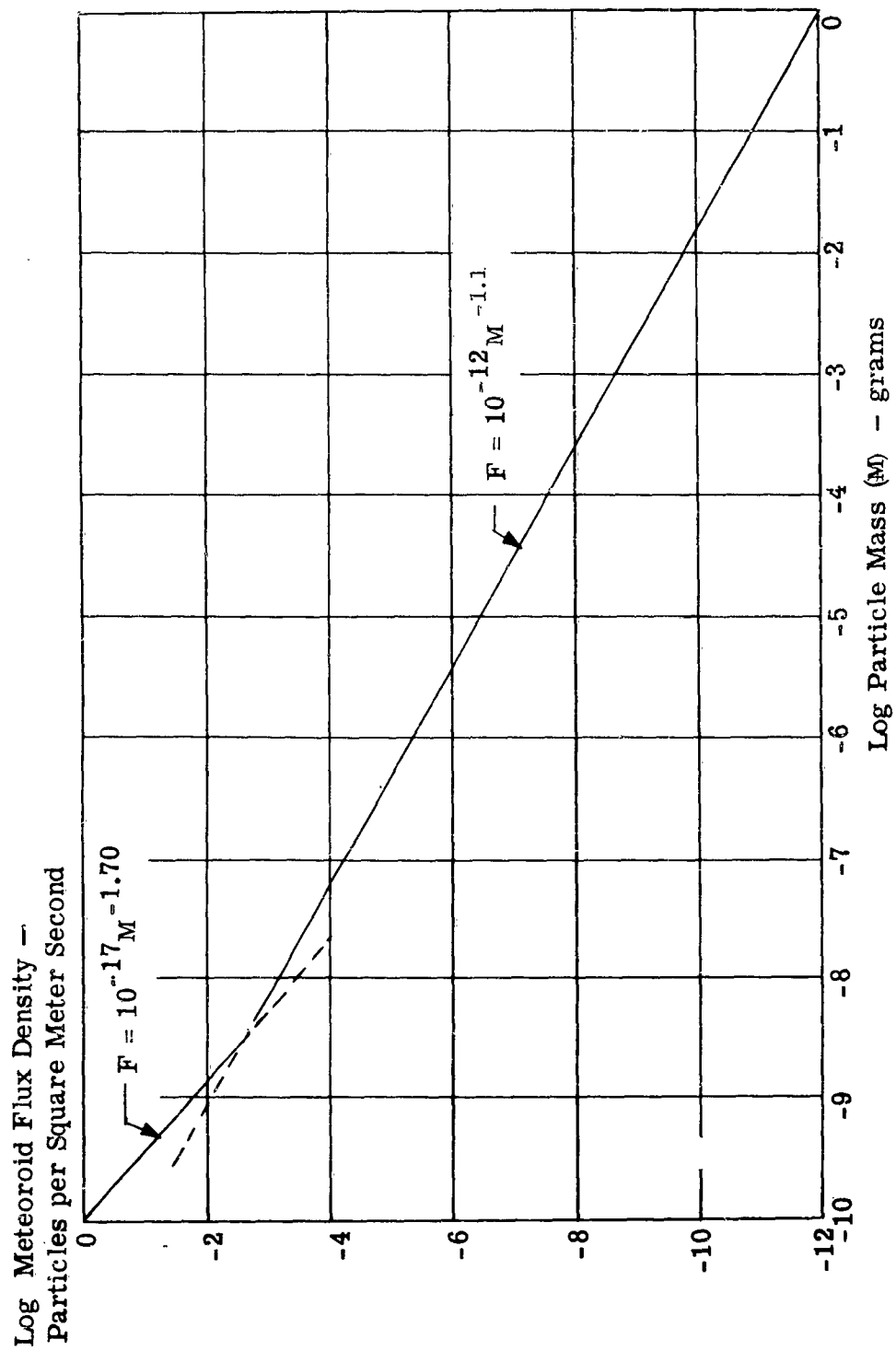


Figure 23. Meteoroid Flux Density

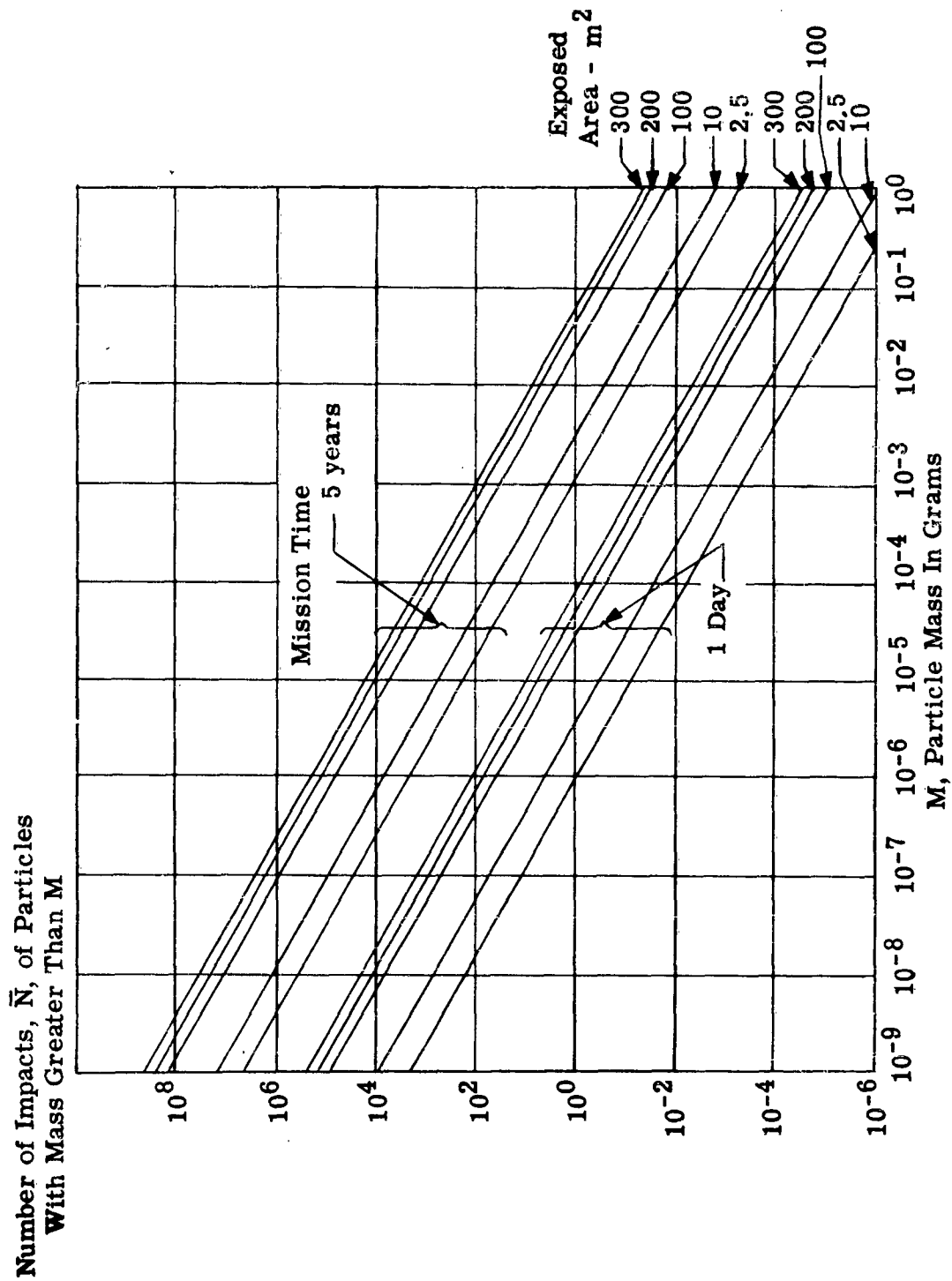


Figure 24. Micrometeoroid Impacts as a Function of Mission Time and Exposed Area

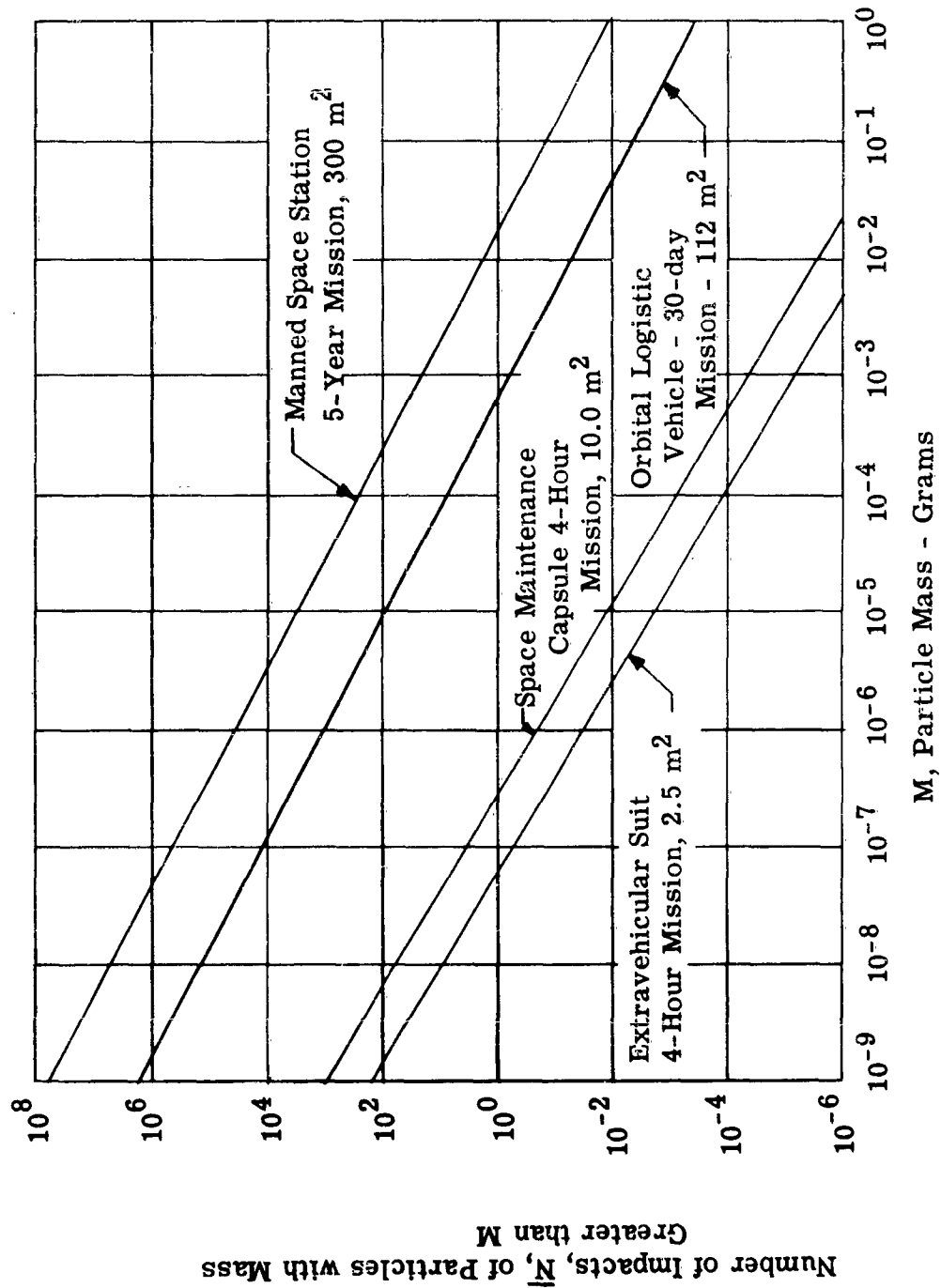


Figure 25. Micrometeoroid Impacts During Missions by Various Vehicles

of the above energies. If this residual energy is large enough, the penetrating particle may be hazardous not only from the standpoint of decompression of the protective enclosure, but also from the possibility of inflicting an injury to the operator. In addition, spalling from nonpenetrating particles might also inflict damage to the components and occupants as well as increasing the probability of wall failure under pressure. Therefore, an adequate design of the vehicle wall must give consideration to the minimization of micrometeorite penetration and possible spalling.

a. Penetration and Cratering

An extensive survey of all available theoretical and experimental work on hypervelocity impact phenomena up to September 1961 has been compiled by Hermann and Jones⁽³²⁾. Experimental data in the velocity range above 6 km/sec up to the time of this survey were scant, but more studies are presently being performed at higher velocities. The penetration equation, derived by Bjork^(30,33) based on a numerical solution of two-dimensional hydrodynamic motion equations, relates penetration with the 1/3 power of momentum. This equation has been adopted as the best approach to date for prediction of cratering and penetration phenomena. Under the assumption that this 1/3 power law holds for the entire velocity range of micrometeoroids, an extrapolation of the experimental data (measured at lower velocities) to velocities above 6 km/sec can be made.

The Bjork penetration equation is stated as:

$$p = k_1 (M V)^{1/3} \quad (3)$$

where p is the penetration in centimeters, M is particle mass in grams, and V is particle velocity in kilometers per second. The constant k_1 has values 1.09 and 0.606 (not dimensionless) for aluminum on aluminum and iron on iron, respectively. It can also be assumed that thin target penetration is 1.5 times that of thick or semi-infinite targets.

b. Spalling

In spite of the protection afforded by a single solid wall, a hazard arises from those impacts which do not penetrate but cause some of the material to break off the inner wall. The kinetic energy may be sufficiently high to do considerable damage. The breaking off is called spalling and is attributed to a strong compression wave which travels through the wall. The stress created on the inner wall is a function of the distance from the point of impact and the strength of the material. The wall thickness required to prevent spalling can be determined as a function of surface area, mission time, and meteor impact frequency in a manner similar to that of penetration.

Two methods of reducing spalling and penetration effects may be considered. First, spalling is a function of material strength⁽³⁴⁾. Hence, the thickness at which spalling takes place may be appreciably reduced by the use of tough steel with

a strength three times that of tough aluminum. Another method is to break up the meteoroid particle into several fragments using a bumper. Also, the use of a multi-layer structure will reduce the transmission of the (pulse) waves which produce spalling. Both means will result in either a decrease of the penetration and spalling probability (at constant wall thickness) or will allow a reduction of the wall thickness (at constant probability). Meteorite bumpers are discussed in more detail in a later section of this chapter.

5. Penetration and Spalling Probability Calculations

a. Penetration

If it can be assumed that thin aluminum wall target penetration is 1.5 times that of thick, or semi-infinite targets, the use of Bjork's calculation to obtain skin thickness, t_s , results in the equations:

$$t_s = 10^{-3.33} F^{-0.196} K V^{0.333} (10^{-2.568} \leq F \leq 1.0) \quad (4)$$

$$t_s = 10^{-3.6} F^{-0.3} K V^{0.333} (F < 10^{-2.568}) \quad (5)$$

where $K = 1.5 K_1$

Conversely, given the thickness, one may obtain the penetrating flux from:

$$F = 10^{-17} t_s^{-5.10} K^{5.10} V^{1.70} (10^{-10} \leq m \leq 10^{-8.5} \text{ g}) \quad (6)$$

$$F = 10^{-12} t_s^{-3.33} K^{3.33} V^{1.11} (m > 10^{-8.5} \text{ g}) \quad (7)$$

where F is the penetrating flux density in number of penetrations per square meter and second of a target, t_s is cm of thickness, and V is particle velocity in kilometers per second.

A value of $F = 10^{-2.568}$ or 2.704×10^{-3} penetrations/ m^2 /sec represents the point of intersection of the curves of Equations (1) and (2) as shown in Figure 23. For a nominal impact velocity of 40 km/sec and an aluminum target, the skin thickness becomes 8.3×10^{-3} cm. Most skin thicknesses will exceed 10^{-3} cm. Thus, for the meteorite velocity and meteorite density considered in this analysis, only larger particles will play a role. Hence, the Equation (4) and (7) are adopted for convenience. We can now rewrite the two equations for aluminum skin and for a nominal particle velocity of 40 km/sec

$$t_s = 10^{-2.851} F^{-0.3} \text{ (cm)} \quad (8)$$

$$F = 10^{-9.504} t_s^{-3.33} \text{ (n/m}^2\text{-sec)} \quad (9)$$

For a calculation of penetration probability, the Poisson distribution for the probability of any number of encounters is used. Let P_n be the probability of one or more penetrating encounters and \bar{N} the expected number of penetrating encounters in the course of the mission. The assumption of a Poisson distribution implies that P_0 , the probability of no encounters, is

$$P_0 = e^{-\bar{N}} \quad (10)$$

and

$$P_n = 1 - P_0 = 1 - e^{-\bar{N}} = - \sum_{i=1}^{\infty} \frac{(-1)^i}{i!} (-N)^i \cong \bar{N} \quad (11)$$

The approximation is valid for small values of \bar{N} , since N^{-2} and higher power terms of the expansion may be neglected. For survival probabilities of 0.90 or more, the neglected terms are practically insignificant.

Figures 26 and 27 present the relationship between penetration flux and aluminum target thickness. The penetrating flux density and the number of penetrating encounters are related by the equation:

$$F = \frac{\bar{N}}{A \tau} \quad (12)$$

The parameter \bar{N} is the mean number of meteoroid impacts which are able to penetrate a structure. Again F is expressed in number of penetrations per m^2 -sec; τ denotes the mission time in seconds and A the exposed area in m^2 .

According to an estimate recently communicated by Goddard Space Flight Center⁽³¹⁾, the depth of penetration is approximately equal to the radius of the hole produced, so that the minimum area of a perforation may be calculated from the relationship

$$A_p = \pi p^2 \quad (13)$$

From the minimum hole area and the expected number of penetrations, the minimum total area of the perforations may be obtained.

$$A_p \sum = \bar{N} \pi p^2 \quad (14)$$

Since the calculation of the constants K , k_1 requires a numerical solution of the equations for each combination of velocity and material, a general method covering a large number of structural materials, consistent with Bjork's approach, was considered in order to compute minimum skin thickness required for an acceptable penetration probability. One such method is suggested⁽³⁵⁾, in which skin thickness is related to the

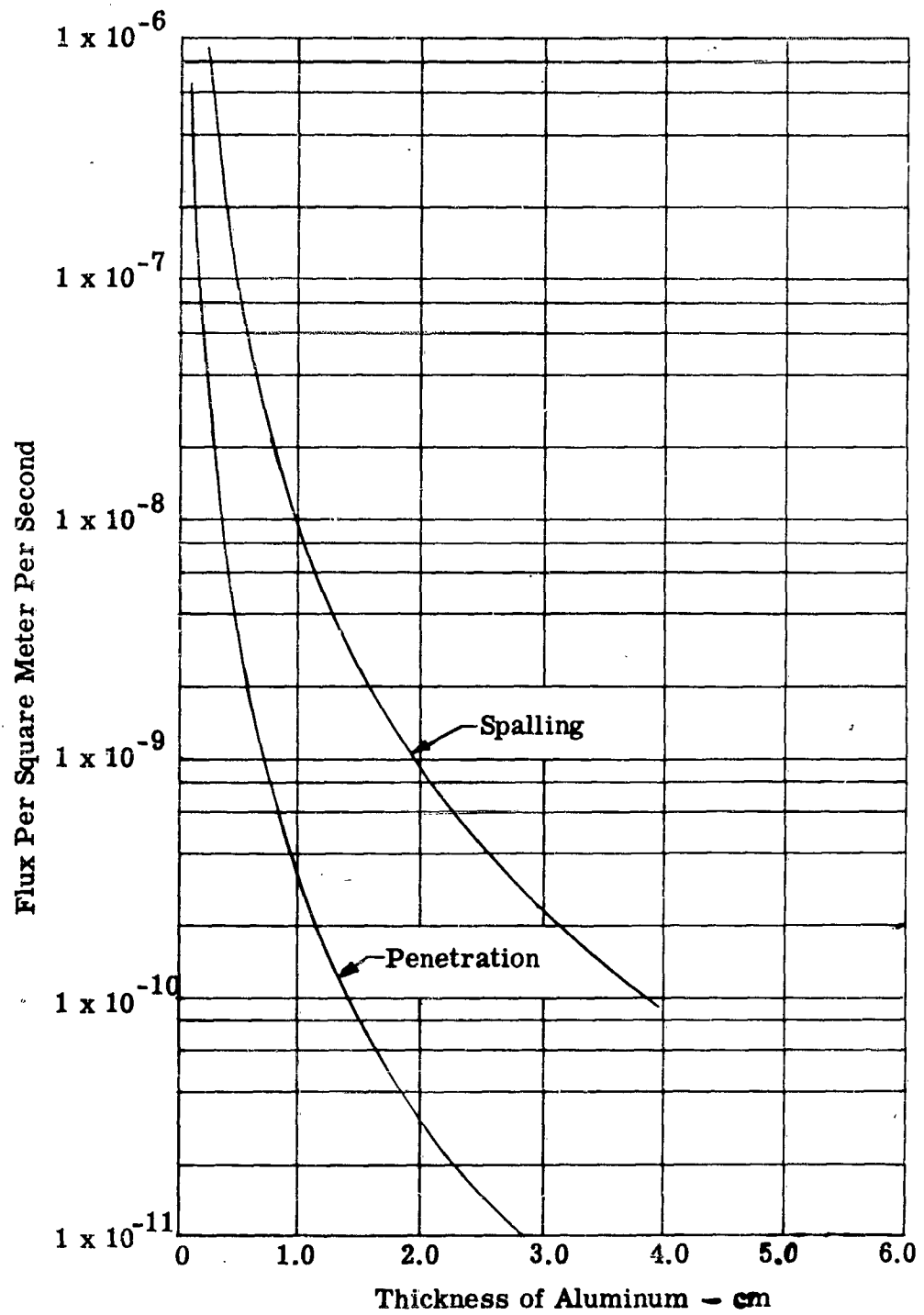


Figure 26. Penetration and Spalling Flux versus Thickness of Aluminum
Velocity = 40 km/sec

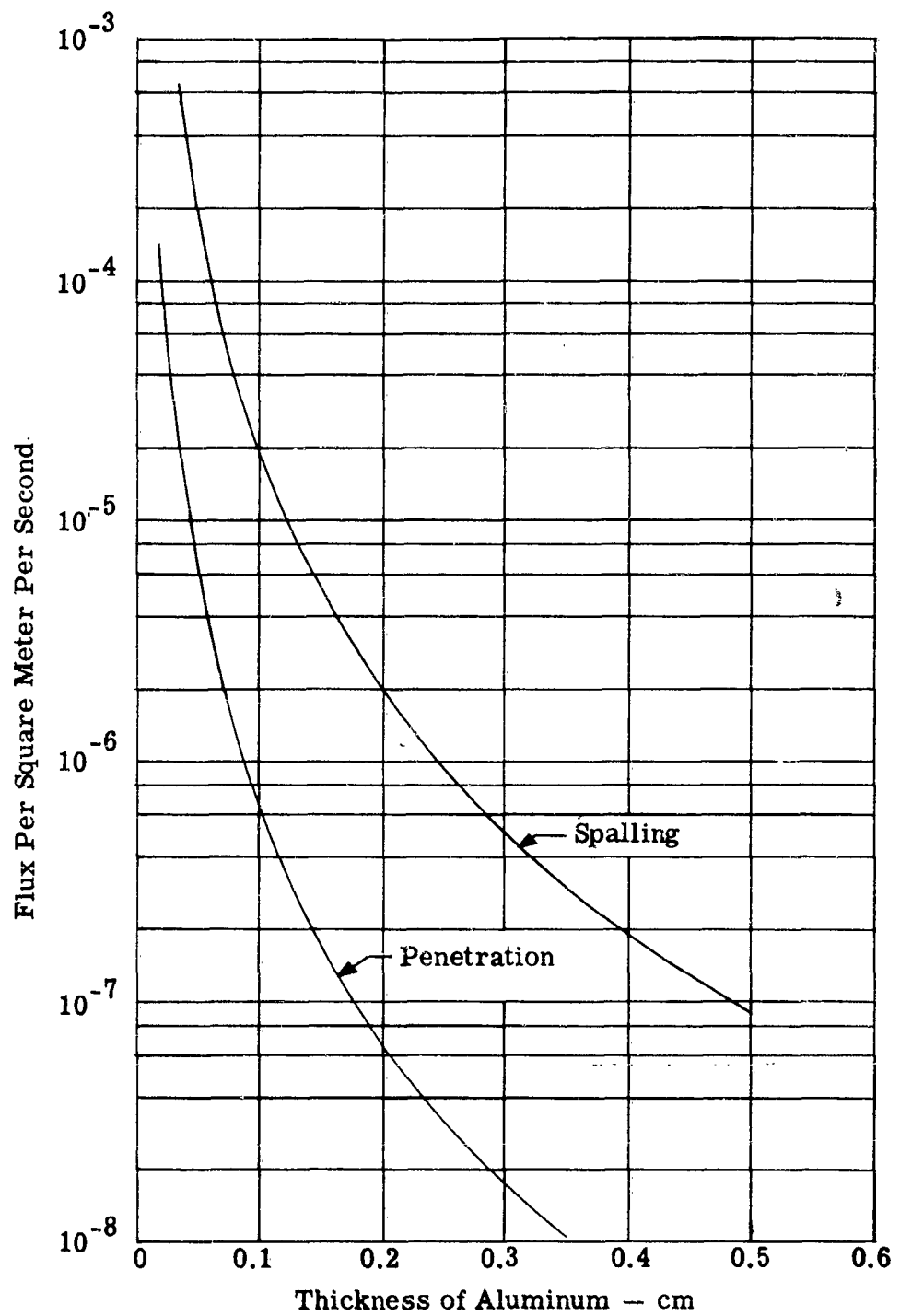


Figure 27. Peretration and Spalling Flux versus thickness of Aluminum
Velocity = 40 km/sec

particle flux F ; the flux density constants α and β ; an empirical constant γ ; the particle velocity V ; the particle density ρ_p ; the target density ρ_T ; velocity exponent θ ; density ratio exponent ϕ ; target area A ; mission period τ ; and modulus of elasticity of the target material E_T . From these quantities, the skin thickness is computed:

$$t_s = 1.24 \alpha^{1/3} \beta V^\theta \left[\gamma \frac{(\rho_p / \rho_t)^{\phi-1/3}}{T^{1/3-\theta/2} E_T^{\theta/2}} \right] \left(\frac{A \tau}{N} \right)^{1/3 \beta} \quad (15)$$

the particle flux density and the constant α and β are determined from Equations (1) and (2). The empirical constant γ can be assumed not to vary with the material and is conservatively estimated to be 3.5. Average or maximum values for particle velocity and density may be used. The mission time, vehicle surface area, and target material data for each vehicle and mission must be given. Values for θ and ϕ are each assumed to be $1/3$, in accordance with the best estimates available on hypervelocity impact data. It is also assumed that the penetration is a function of $v^{1/3}$, in accordance with Bjork's prediction.

It can be seen from Equation (15) (after introducing the values suggested) that thickness becomes a function of the mission duration and two material properties; target density and modulus of elasticity.

It is now possible to compute an "equivalent aluminum thickness" for many target materials by:

$$t_{s_x} = t_{s_{al}} \left[\frac{\rho_{T_{al}} E_{T_{al}}}{\rho_{T_x} E_{T_x}} \right]^{0.1667} \quad (16)$$

where t_{s_x} = thickness of target material x .

E_{T_x} = Modulus of Elasticity of material x .

ρ_{T_x} = Density of material x .

This equation enables the estimation of the skin thickness of various materials required to provide a desired degree of protection against meteorite penetration for any length of time in the vicinity of the earth.

b. Spalling

Based on the method of Reference (34) and on the Equation (1), the following relationship between spalling probability and wall thickness has been derived for a (nominal) meteoroid impact velocity of 40 km/sec and for aluminum targets.

$$D = 10^{-2.41} F^{0.3} \quad \left[\text{cm} \right] \quad (17)$$

or

$$F_S = 10^{-8.04} D^{-3.33} \quad \left[\text{n/m}^2\text{-sec} \right] \quad (18)$$

In these equations, F_S is the meteoroid spalling frequency similar to that used previously for penetration and expressed in the same units as F and D is the thickness of aluminum required to resist spalling for the given flux density. The relationship is plotted in Figures 26 and 27 and may be used to estimate spalling in the same manner as Equations (8) and (9) are used for penetration.

The ratio of thicknesses required to prevent spalling to that required to prevent penetration is independent of the flux density F_S . This may be seen from Equations (8) and (17).

$$\frac{D}{t_s} = 10^{0.441} = 2.74 \quad (19)$$

This ratio may be applied in lieu of separate computations of spalling effects.

c. Application to Space Vehicles

(1) Vehicle Skin Properties

The equations of the foregoing section are valid for a solid wall of density ρ_T . However, the structure of a space system consists of several layers of material of different densities and moduli of elasticity, and the impact phenomena will differ from that of a single wall of uniform density. Since no experimental data are available, at present, on multi-wall penetration and spalling, it was necessary to develop techniques for approximating the probability of the occurrence of these phenomena. As a first approach, the average density ρ_T was used. Each structural layer was weighted by its density and thickness and the average density of the multi-wall structure was computed. This value was then substituted into Equation (16) to obtain the equivalent aluminum thickness. It must be emphasized that the penetration probabilities computed by the above described technique are high, relative to those which will be actually experienced by the vehicle systems.

Table XIII shows the densities of components representative vehicle walls and the computed average densities of the composite wall. These values may be compared with that of aluminum, which is approximately 0.1 lb/in.³. The average wall densities range from about 1 to 1.5 times the density of aluminum, with the exception of the nylon/rubber space suit.

TABLE XIII
AVERAGE DENSITY AND MODULUS OF ELASTICITY
OF VEHICLE WALL MATERIALS

VEHICLE TYPE	Material Layer Thickness cm	Density ρ , lb/in. 3	Average Density ρ , lb/in. 3	Modulus of Elasticity (psi)
SPACE LOGISTIC VEHICLE (Double Wall)	Haynes No. 25/0.011	0.330		34.2×10^6
	Haynes No. 25/0.11	0.330		34.2×10^6
	Inconel/0.005	0.309	0.155	31.0×10^6
	Insul. Inconel/0.005	0.309		31.0×10^6
	Al No. 2014/0.102	0.100		10.7×10^6
SPACE STATION	Aluminum/0.160	0.111	0.111	10.7×10^6
SPACE MAINTENANCE CAPSULE	Corrugated Aluminum/0.041	0.100		10×10^6
	Aluminum Tubing/0.050	0.100		10×10^6
	Magnesium/0.025	0.168	0.081	6.5×10^6
	Resin/0.050	0.061		not available
	Magnesium /0.010	0.068		6.5×10^6
SPACE SUIT	Nylon/0.012	0.041		
	Nylon or Dacron/0.036	0.041	0.041	0.56×10^6
	Nylon or Dacron/0.018	0.041		

Moduli of elasticity for all vehicles (Maintenance capsule and space suit not included) range from about 10×10^6 to 30×10^6 lb/in.²: i.e., they represent values of from 1 to 3 times that of aluminum. The equivalents of density and Young's modulus of the vehicle walls are greater than that of aluminum (space suit excluded). The moduli are 3 times as high and the densities 1.5 times. Consequently, the thickness of these walls will be less than that of solid aluminum walls for the same penetration probability: $\left(\frac{1}{(1.5)(3)} \right)^{1/6} = 0.779$. See Equation (16). The space suit is treated separately; its skin is not metallic and has different properties. Also, since the wall properties of the capsule are approximately those of aluminum, the wall will have the penetration probability of a solid aluminum wall 0.9 times the total thickness of the composite capsule wall.

In conclusion, we can state that the use of Bjork's method and data for aluminum targets and particles will result in values which do not differ very appreciably from those of the various other materials. The thickness or penetration probability values obtained would be at least conservative. Hence, Equation (4) may be used to obtain aluminum skin thickness at a given penetration probability. For materials other than aluminum such as those of the Bell Double-Wall vehicle listed in Table XIII, a factor of 0.78 must be applied to the thickness obtained from Equation (14). For the Double-Wall, this multiplier is 0.78 to account for the difference in material properties. For nylon, however, this factor is 1.87 because of the lower density and a lower modulus of elasticity.

The multi-layered walls will reduce the effect of spalling considerably in a manner analogous to the penetration reduction.

Typical flux values for penetration and spalling are shown in Table XIV for several vehicles, a space maintenance capsule, and a space suit. The space suit consists of several layers of nylon skin. It was assumed that nylon will behave similarly to the metal skin used in the other space vehicles and hence the equivalent aluminum skin thickness was calculated from Equation (16). This assumption was made, even though it is not known, how synthetics will behave under impact because no data are available which describe the behavior of such materials subjected to hypervelocity impacts.

It can be concluded from Table XIV that for a 4-hour mission and a surface area of 10 square meters, a survival probability of approximately 0.9 is barely attainable for most of the vehicles without a bumper. Application of a bumper reduces the penetration flux by a factor of about 38 as described in the following section.

(2) Meteorite Bumper Effects

A bumper or shield is a thin sheet of material arranged outside the vehicle, yet fairly close to the skin. An impinging particle would break up upon collision

TABLE XIV
PENETRATION AND SPALLING PARTICLE FLUX FOR
SEVERAL SPACE VEHICLE STRUCTURES

Vehicle	Wall Thickness t_s, c_m	Equivalent Aluminum Thickness t_s, c_m	Flux Penetration per square meter and second	Flux of Spalling Particles per square meter per second	No. of Penetrating Encounters, N, per 10 square meter surface area for 10-day mission	No. of Penetrating Encounters, N, per 10 square meter surface area for 4-hour mission
Space Logistic Vehicle (Double Wall)	0.13	0.17 (1)	1.1×10^{-7}	3.2×10^{-6}	0.95	0.016
Space Station	0.16	0.16	1.4×10^{-7}	4.0×10^{-6}	1.2	0.020
One-Man Maint. Capsule	0.19	0.17 (4)	1.1×10^{-7}	3.2×10^{-6}	0.95	0.016
Space Suit	0.066	0.035 (2)	220×10^{-7}	633×10^{-6}	47.5 (3)	0.79 (3)

NOTES: 1. Thickness x 1.28
2. Thickness x 0.534 for Nylon
3. Per 2.5 square meters of exposed area
4. Thickness x 0.90

with the bumper, as proven by experiments. Properly designed space shields offer structural weight saving because a smaller total skin thickness is required for the same penetration probability as for a solid wall.

Fragments of a meteorite have less mass and, consequently, reduced penetrating capability than the original particle. By proper spacing, the stream of particles is spread over a large area of the main skin; thus, the probability of closely spaced impacts is reduced and penetration will become less likely. The number of fragments is of importance because the penetration is proportional to the $1/3$ power of the mass of particles of equal velocity. Thus, two halves of a particle would penetrate $(1/2)^{1/3} = 79\%$ the depth of one particle, three thirds 69% etc., if we assume that the velocity of the fragments is equal to that of the unbroken particle. The average penetration depth is assumed to be about 33% with the presence of a bumper⁽³⁴⁾. Hence, for the same penetration probability, the total thickness of the main plate plus bumper or outer shield is reduced to $1/3$ the thickness of a solid wall alone. In a similar manner the bumper reduces the spalling probability.

It must be emphasized that little information is available on bumper design. Application of the technique described below to actual skin design must be considered tentative, awaiting confirmation by measurements on the impact of high velocity particles. Such experiments are currently scheduled. The method suggested in Reference 34 is summarized below in order to give some indication of the results which might be obtained by a proper wall design.

We will assume that the main plate thickness will be only $1/3$ the thickness required for a solid wall and that an aluminum particle disintegrates after hitting a bumper whose thickness is $1/10$ the particle diameter. Reference 34 states that aluminum particles shatter at collision velocities of 6 to 20 km/sec, while glass spheres break up at 2 km/sec. Since the minimum penetrating particle diameter d of a hypersonic particle is related to the solid plate thickness t_s by the relationship

$$t_s = 1.5 d, \quad (20)$$

and since

$$t_b \geq 1/10 d, \quad (21)$$

the bumper thickness required may be obtained as a function of the solid skin thickness calculated for the desired penetrating flux

$$t_b \geq (1/15) t_s \quad (22)$$

for a solid wall. The proper spacing is important and is suggested by Jaffee and Rittenhouse⁽³⁴⁾ to be

$$s = 2 t_t \sqrt{\frac{t_t}{t_b \rho_b}} \quad (24)$$

The spacing is represented by s , bumper thickness by t_b , main plate thickness by t_t in cm, and bumper density by ρ_b (gm/cm^3).

The penetration and spalling fluxes for the space vehicle structure presented in Table XV were computed using the material thicknesses found for a bumper-free design, and by taking into account that by adding a suitable bumper, the wall thicknesses become equivalent to three times their actual value. These values are much more representative of the actual penetration probabilities experienced by vehicles than those found in Table XIV and clearly point out the increase in system survivability which results with the employment of a micrometeorite bumper. Figure 28 presents the penetrating flux as a function of skin thickness, with and without a bumper, for a 4-hour mission with a space maintenance capsule. Once again, the benefits derived from the employment of a spaced bumper structure are clearly obvious. It is important to note that a similar decrease in penetration probability will result if the extravehicular suit were designed with a spaced metallic structure. However, to design the suit in such a manner would result in an extremely stiff and unwieldy system which would considerably decrease the capability of the maintenance worker to engage in his required task. This problem is discussed in greater detail in Chapter 6.

6. Effects After Impact

In the foregoing analyses, the least number of penetrations of particular vehicles have been computed. We shall now study what the effect of the penetrations on the state of the cabin gas will be. One may safely state that not all the penetrations are catastrophic in the sense that vehicle and/or occupant become damaged beyond repair.

If the vehicle wall is penetrated, the cabin gas will immediately suffer a change of state, the rate of which will depend upon the vehicle volume, initial temperature, initial pressure and the effective hole diameter. The discharge may be assumed to be adiabatic if the effective hole diameter is large, or isothermal if the hole diameter is small. These two extreme changes of state are the limits of probable polytropic changes of state within the perforated enclosure. The adiabatic discharge may be much more critical to the vehicle occupant because of its fast rate, but much less frequent because of the low flux density of large particles. The isothermal case is much less dangerous (per penetration) but more frequent; however, because of the higher frequency of small penetrations, it may well represent the case of greater importance.

Gell⁽³⁷⁾, presents data which indicate quite unexpected results following particle penetration. In these studies, an aluminum pressurized vessel with a 0.070 inch wall housed in a vacuum chamber at 1.75×10^{-1} mm Hg, was impacted by

TABLE XV
PENETRATION AND SPALLING PARTICLE FLUX FOR SPACE
VEHICLE STRUCTURES WITH BUMPER

Vehicle	Wall Thickness t_s, c_m	Equivalent Aluminum Thickness t_s, c_m	Flux Penetration per square meter and second	Flux of Spalling Particles per square meter per second	No. of Penetrating Encounters, N, per 10 square meter surface area for 10-day mission	No. of Penetrating Encounters, N, per 10 square meter surface area for 4-hour mission
Space Logistic Vehicle (Double Wall)	0.13	0.17(1)	3.0×10^{-9}	0.86×10^{-7}	2.6×10^{-2}	4.3×10^{-4}
ESWS Space Station	0.16	0.16	3.6×10^{-9}	1.1×10^{-7}	3.1×10^{-2}	5.2×10^{-4}
Remora ⁵	0.19	0.17(2)	3.0×10^{-9}	0.86×10^{-7}	2.6×10^{-2}	4.3×10^{-4}

NOTES: 1. Thickness x 1.28
2. Thickness x 0.90

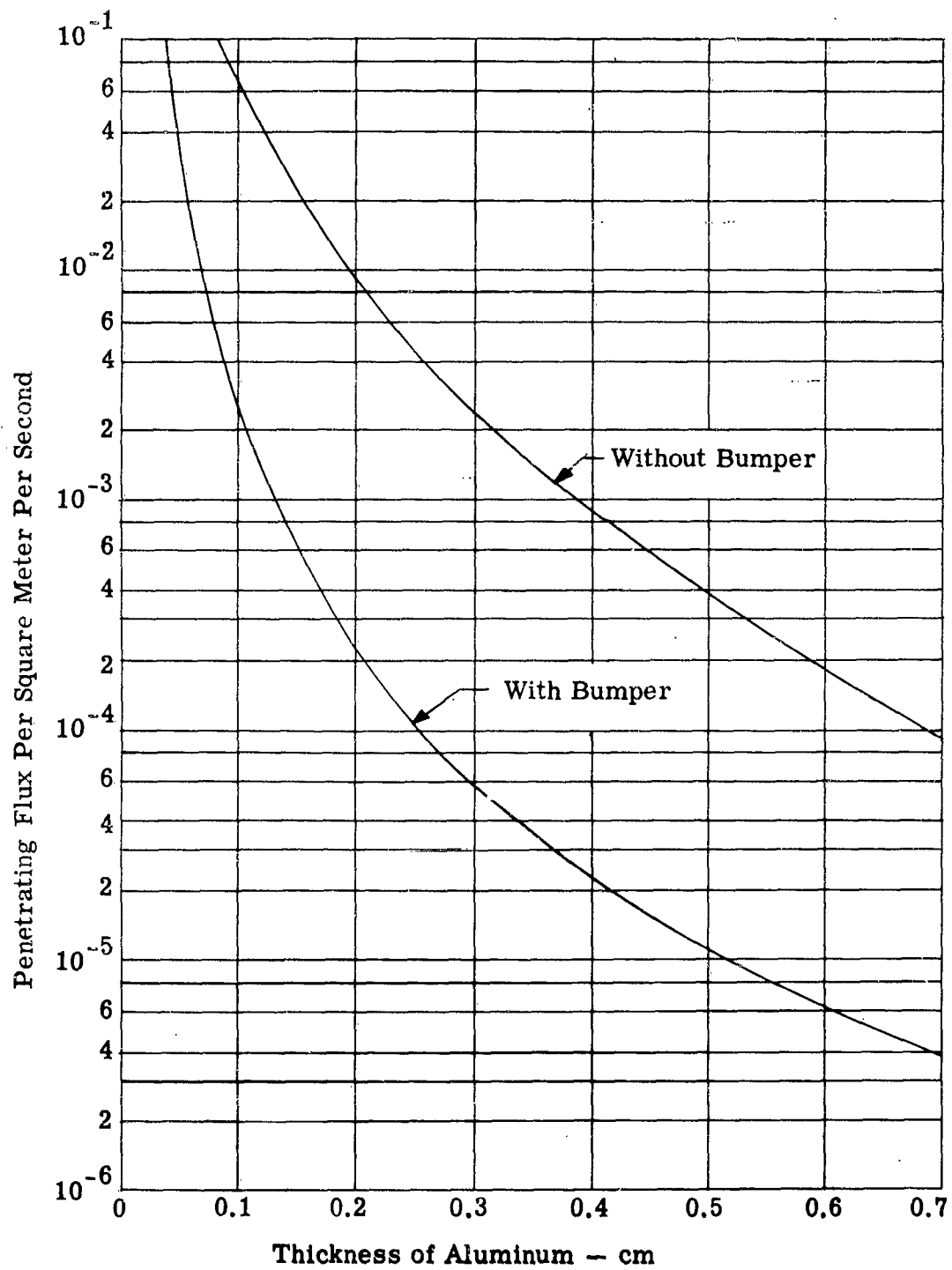


Figure 28. Penetration Flux for a 4-Hour Space Maintenance Capsule Mission

aluminum particles ranging in diameter from 2 μ to 2 mm. The particle velocities were approximately 8 km/sec for the large particles and 20 km/sec for the very small particles. Upon penetration, the particle and the aluminum wall melted and/or vaporized, and upon entering the atmosphere resulted in a rapid explosive type oxidation. The intensity of the oxidation was demonstrated to be a function of the O_2 partial pressure. However, even in the case where sea level pressure and composition were employed, the effects of the oxidation were severe enough to cause minor skin injuries and shock to the animal in the cell. It is apparent from these results that a serious potential danger over and above that predicted by the standard expansion laws might exist when a vehicle is penetrated. The effects of the explosive oxidation, the severity of which is certainly a function of vehicle volume, may result in serious damage to the occupant and the temperature and pressure sensitive space vehicle components.

The effects of penetration are studied in four steps; energy transfer between the meteorite and the vehicle wall or cabin gas are not considered in the first two analyses. The effects of an instantaneous isochoric energy transfer from the meteorite to the cabin gas are computed in the third step, while the deceleration by the cabin gas of a spherical, constant and nonreacting mass is studied in the fourth step. The last two steps may be used for the interpretation of the results of Gell's experiments. The role of the initial free volume of the penetrated enclosure is substantiated by the analyses, and it was also shown that a nonreacting, constant mass particle may not transfer enough energy to the gas to cause the untoward effects described by Gell.

The first two steps of the analyses, the isothermal and the adiabatic changes of state within the enclosure, assume that no disturbing transients affect the bulk of the cabin gas between puncture and start of the discharge.

a. Isothermal Discharge

The change of state within the vehicle is considered to be isothermal if the discharge time is long and it can be assumed that the life support system remains functional and can maintain constant temperature. Collisions with small particles causing at least initially isothermal efflux may be considered to be the most probable. The isothermal change of state is described by:

$$\frac{P}{P_o} = \exp \left[\frac{-A}{V_o} t \sqrt{g \gamma k R T_o} \right] \quad (25)$$

where: $k = \left(\frac{2}{\gamma + 1} \right)^{\frac{\gamma}{\gamma + 1}}$ (25a)

where P the enclosure pressure at time t after perforation
 P_0 pressure before perforation
 V_0 free air volume of the enclosure
 A effective cross section of the puncture
 γ adiabatic exponent
 g standard gravity
 R specific gas constant
 T_0 air temperature in the enclosure

Equation (25), with O_2 as the pressurizing gas, becomes:

$$\frac{P}{P_0} = \exp \left[\frac{-A}{V_0} t \sqrt{27 T_0} \right] \quad (26)$$

Assume that the vehicle pressure, prior to penetration, is 5.0 psia (258 mm Hg). Further assume that the low pressure limit for sustaining life for a short duration is 120 mm Hg. The effective diameters of the perforations which cause this pressure of 120 mm Hg to be reached 5 and 30 minutes after penetration have been computed for an extravehicular suit of 2.5 m² exposed area and a one-man maintenance shuttle capsule of 10 m² area. The free air volume in the suit was estimated to be 1.75 ft³ and in the capsule to be 21 ft³. The initial temperature was set at 75°F. Assume further that the smallest diameter of the puncture is 1.5 times the diameter of the penetrating particle. Equation (26) leads to the following relationship between free volume of the enclosure, discharge time and volume (V_p) of the penetrating particle:

$$V_p = 0.273 \sqrt{\frac{V_0^3}{t}} \text{ cm}^3 \quad (27)$$

Using the particle volume (V_p) and the particle density (ρ_p), the particle masses may be computed by the following formula:

$$m_p = V_p \rho_p \quad (28)$$

where $0.05 \text{ g/cm}^3 < \rho_p < 7.0 \text{ g/cm}^3$

Table XVI presents the estimated particle masses.

TABLE XVI

PARTICLE MASSES CORRESPONDING TO HOLE DIAMETERS FOR
A GIVEN RATE OF PRESSURE DROP (MASS IN MICROGRAMS)

	PARTICLE DENSITY (ρ_p)					
	0.05 g/cm ³		2.7 g/cm ³		7.0 g/cm ³	
$t \backslash V_o$	1.75	21.0	1.75	21.0	1.75	21.0
300	6.08	254	341	13710	850	35,500
1800	0.414	17.3	22.3	934	57.9	2420

V_o = free volume, ft³

t = discharge time, sec

The occurrence of masses greater than any one of these listed masses (m_p) is, roughly spoken, inversely proportional with m_p . A smaller mass may, however, be much more brittle than a larger one of the same volume. Consequently, it may break up more easily upon contact either with a bumper or with a wall of the enclosure. It is conceivable that even in the absence of a bumper, fragmentation of brittle masses may reduce their penetration probability.

The frequency of impacts with particles heavier than m_p was computed using the frequency function presented in Figure 23. The results of these computations yielded estimates of the longest time intervals between subsequent impacts of particles, the mass of which exceeds m_p . Table XVII presents a summary of these calculations. Investigation of this table shows that larger enclosures are safer than smaller ones, in spite of their greater exposed surface. Further, reading the columns points out that encounters with dense particles are much less frequent than encounters with less dense ones. Roughly, 20 times more encounters will occur with particles causing dangerous discharge within at least 30 minutes than with meteorites large enough to cause the same discharge within 5 minutes. One should be reminded here that the brittleness of lighter particles ($\rho_p \leq 1 \text{ g/cm}^3$) may reduce their penetration probability. We shall neglect this possible reduction in the following estimation of the cumulative effect on the cabin pressure of a number of small holes.

TABLE XVII

**ESTIMATED TIME INTERVALS BETWEEN
CRITICAL PENETRATIONS (Days)**

$t \backslash V_o$	PARTICLE DENSITY (ρ_p)					
	0.05 g/cm ³		2.7 g/cm ³		7.0 g/cm ³	
	1.75	21.0	1.75	21.0	1.75	21.0
300	7.31	104	51.8	1020	1845	26,148
1800	0.37	5.22	3.0	48.4	90	1311

V_o = free volume, ft³

t = discharge time, sec

The greater number of small holes will also offer escape paths to the cabin or suit atmosphere. The effect of N small holes is found to be expressed by the following equation if it is assumed (1) that the impingements are separated by the same time intervals, and (2) that each particle will cause the same size hole:

$$\frac{P_N}{P_o} = \left(\frac{P_1}{P_o} \right)^{N \frac{N-1}{2}} \quad (29)$$

where

t_1 = time interval between two impingements of small particles

P_1 = pressure after discharging through the first small hole at the time t_1

N = the number of impingements

P_N = the pressure just before the N th impingement

The longest time interval between two impingements of meteorites piercing at least a small hole t_1 , is found from Table XVII. For the space suit, this time is 0.37 days, or 33,000 seconds. P_1/P_o was computed from Equation (26) and found to be 6.6×10^{-8} . Consequently, the drainage between encounters with two small particles is enough to make the cape useless. A more detailed analysis would take into account that not all of the "smaller" particles are of the same size.

b. Adiabatic Discharge

The case of the adiabatic discharge is studied rather briefly because it is less likely to occur than the isothermal change of state. The change of state of the cabin gas will deviate from the isothermal case and approach adiabatic if the expansion is so fast that the life support system, or any other heat source, cannot release energy fast enough to compensate for the expansion work. A fast discharge requires a large effective discharge cross section and will not be a frequent phenomenon. Were the life support system damaged, even with a small opening, the change of state would then move slowly from the initially isothermal to the finally adiabatic condition as the heat capacity of the originally "live" occupants becomes exhausted.

The following equation describes the change of state of the cabin gas after decay of any starting transient if neither a gas supply nor a heating source exist and the gas velocity in the exit opening is sonic.

$$\left(\frac{P_o}{P} \right)^{\frac{\gamma-1}{\gamma}} = \frac{T_o}{T} = \left[1 + \frac{A}{V_o} c_o \frac{\gamma-1}{2} \left(\frac{2}{\gamma+1} \right)^{\frac{\gamma+1}{2(\gamma-1)}} \right]^2 \quad (30)$$

where,

P	pressure in the cabin
T	absolute temperature of the gas in the cabin
A	effective area of the perforations
V _o	initial gas volume
c _o	sonic velocity in the gas at the beginning of the outflow
γ	isentropic exponent
t	time

(Quantity without subscript: instantaneous value, subscript "o" refers to values at the beginning of the expansion.)

The above equation again shows the importance of the initial volume from which the expansion takes place, the roles of the isentropic exponent and molecular weight of the cabin gas, the initial temperature, and the hole size. We shall use fairly large, though rather infrequent, meteorites in our numerical evaluation. Particles exceeding 23.5 mg are encountered at the rate of 6.3 (10⁻¹¹)/sec, m². Such a particle, if spherical and consisting of aluminum, has a diameter of 0.127 cm and will result in a perforation of 0.191 cm throat diameter. The smallest time intervals between encounters with meteorites of at least this mass (23.5 mg) with the extra-vehicular suit (2.5 m²), a maintenance capsule (10 m²), and a vehicle (1000 m²), are 74,000, 18,400, and 184 days, respectively. The time for pressure

reduction from 5 to 2.4 psia was also computed for the space suit. One obtains, using Equation (30) that pressure drop, leading to the critical 120 mm Hg. cabin pressure, is reached within five seconds, the time being proportional to the volume of the perforated enclosure. The adiabatic temperature ratio, corresponding to the pressure drop from 5 to 2.4 psia is, 1.233; i.e., the cabin gas will cool from +75 to -24 F. It is reasonable to assume that the change of state will be close to adiabatic.

c. Instantaneous Energy Transfer from a Particle to the Cabin Gas

Energy transferred from a particle to the gas can come only from a chemical or physical reaction between the particle and the gas and from deceleration of the particle by the gas. Interactions between the particle and the vehicle wall and gas, and leaks to the outside are neglected in the following analysis.

The maximum transferable energy is Q_t :

$$Q_t = Q_K + Q_C \quad \text{or} \quad (31)$$

$$Q_t = Q_K \left(1 + \frac{Q_C}{Q_K}\right) \quad (32)$$

where,

Q_K the heat equivalent of the kinetic energy of the particle

Q_C the reaction energy between particle and cabin gas

$$Q_K = \frac{m}{2J} U_o^2 \quad (33)$$

$$Q_C = gm H \quad (34)$$

where,

m mass of the particle

g standard acceleration due to gravity

U_o velocity of the entering particle, after penetration
(depends on the gas composition)

J Joule's constant (mechanical heat equivalent)

Equation (32) will become with equations (33) and (34)

$$Q_t = \frac{m}{2J} U_o^2 \left(1 + \frac{2JgH}{U_o^2}\right) \quad (35)$$

The transferred energy will be mainly of kinetic nature, if

$$\frac{2JHg}{U_o^2} \ll 1 \quad (36)$$

e.g.

$H = 7.25$ Kcal/kg-weight (aluminum and oxygen, the highest possible value)

$J = 427$ mkg/Kcal

$U_o = 15,000$ m/sec

$$\frac{2JgH}{U_o^2} = \frac{6.2 (10^{-2})}{U_o^2} \quad (U_o \text{ in km/sec}) \quad (36a)$$

The contribution of the heat of reaction is already insignificant, from the energy point of view, if U is 1 km/s and if an appreciable fraction of the kinetic energy can be directly transferred to the gas. Assuming total transfer, the temperature and pressure increase in the bulk of enclosed gas follows from the equations below, expressing change of state of constant volume:

$$\Delta T = \frac{Q_t}{C_v G_o} \quad (37)$$

$$\Delta p = \frac{R}{V_o} \frac{Q_t}{C_v} \quad (38)$$

where,

$\Delta T, \Delta p$ the instantaneous isochoric increases of absolute temperature and pressure

Q_t as above, the total heat transferred

V_o original volume

R specific gas constant

C_v specific heat of the gas at constant volume

Table XVIII shows the effect of free volume on transient change of state resulting from energy transfer from a meteoroid. The data were computed for $m = 23.5$ mg mass, $V_o = 15$ km/sec and three values of the free air volume; 1.75 cu ft, 21 cu ft, and 104,000 cu ft. The volumes correspond to that of a shuttle, of a space suit and of a large

vehicle. Initial values of pressure and temperature were 5 psia and 75°F, respectively. The heat transferred is 0.63 Kcal, or 2.52 BTU, and the cabin gas is 100 percent oxygen.

TABLE XVIII
EFFECT OF FREE VOLUME ON TRANSIENT CHANGE OF STATE
CAUSED BY ENERGY TRANSFER FROM A METEOROID

V_o cu ft	ΔT °F	Δp psi
1.75	295	2.77
21	24.7	0.23
104,000	0.0068	4.10^{-5}

The pressure and temperature changes are directly proportional to the kinetic energy transferred from the particles which entered the vehicle and inversely proportional to weight and specific heat of the enclosed gas. The results show that considerable changes can occur in a small volume if significant amounts of energy can be directly transferred to the bulk of the gas. It was shown in the preceding section that the probability of encountering a particle of 23.5 mg mass or greater is small for the space suit and the shuttle. The large station may encounter such a particle, however, the bulk effects on it would not be significant as shown in Table XVIII.

Local values of the change of state would considerably exceed the bulk values. The excess is caused by the difference between the rate at which a hypersonic particle can change the state of its immediate surrounding and the speed at which the changes can propagate into the bulk. It is possible that such local changes could result in the effects noted by Gell⁽³⁷⁾.

The following section will present some considerations on the deceleration of a particle which is subjected to drag forces only. Change of state of the bulk will not be considered.

d. Deceleration of a Particle

Deceleration has been analyzed for two extreme cases, very low and fairly high Reynolds numbers. The undisturbed gas is at the standard condition of the living quarters of a space vehicle. Change of state of this gas, as it interacts with the decelerated particle, was not taken into account. The results justify this simplification. Further assumptions of the analysis are:

- (1) Shape and size of the particle are constant.
- (2) Effect of moderate change of state of the gas along the particle path is included in the drag equation.
- (3) Shock waves are neglected.

The mechanism postulated for the interaction does not lead to significant energy transfer. It is concluded that other hypotheses must be proposed; e.g., deceleration upon impact with a thick wall, or chemical and physical reaction of catalytic nature with the enclosed gas. The reaction might lead to a very great number of small enough decelerable particles; hypersonic or supersonic effects may lead to strong local changes. Such hypotheses can be treated analytically as well as subjected to empirical tests. The results of the original analyses indicated that considerable distances are necessary for direct energy transfer from the particle to the cabin gas. Consequently, we have concluded that insignificant energies are transferred, under the conditions of the analysis to account for the phenomena observed by Gell. The energy transfer treated in the Energy Transfer section is not realizable by the deceleration mechanism described at the beginning of this section. The present study accentuates the need to undergo careful analyses, including the interaction between wake of the meteorite and the meteorite itself, and the transfer of energy from the wake to the bulk gas.

e. Consequences and Problems

The unfavorable effects of a puncture of given size decreases with the size of the initial gas volume. The number of collisions per unit time will, however, increase with the exposed surface of the vehicle. The thickness of the structural wall is determined by size, shape and internal gas pressure, static and dynamic loads. The penetration probability will decrease with wall thickness and with the number, thickness, and arrangement of non-load carrying surfaces which are on the outer side of the structural wall: These may be heat transfer surfaces or intentionally arranged thin shields for the purpose of breaking up impinging meteoroids and hereby reducing their probability of penetration. It appears reasonable to anticipate that a volume exists for any total weight of a space vehicle supporting and nonsupporting surfaces (machinery excluded) for which the pilot's survival probability is highest. The existence of such a volume can be determined after more insight has been gained into the following questions:

- (1) Break up of meteoroids by single and multiple bumpers (effects of material, thickness, spacing, number of bumpers, mass, velocity, material of impinging particle resulting in number, size and velocity distribution, temperature increase, and dispersion of the fragments).
- (2) Energy transfer to the structural wall; further breakup of the fragments.
- (3) Exchange of energy between cabin gas and fragments (shock waves, if any, heat transfer, chemical reactions, change of state).

The questions relative to fragmentation are compounded by the possibility of fragments leaving the cabin and of fragments interacting with self-sealants placed in one or more of the interspaces. The sealant may have additional effects on the heat transfer and on the spalling of the structural wall's inner face.

7. Self-Sealing Repair Methods

A self-sealing vehicle skin, designed according to known principles for stopping fluid leakage after penetration, appears to be a promising method of increasing the survival probability of the vehicle over a long mission time, provided the holes are plugged in sufficiently short time after a particle penetration. A sealant layer incorporated between hard layers of skin could be effective in a vacuum. The heat of the collision may support the sealing process. Furthermore, a permanent patch might be applied to reinforce any damaged skin if the location of the hole is known and accessible.

One approach to such a structure has been subjected to experimental study by the Air Force⁽³⁸⁾. The system consisted of two fluids separated by a membrane. When punctured, the fluids mixed, solidified, and sealed the puncture with relative success. The suggestion was proven when high tensile and high modulus elastomers were used for the walls and for polypropylene oxide-toluene diisocyanate, adduct thickened with colloidal silica and triethylene tetramine. While considerable problems relative to the use of such a design in an operational space vehicle must be resolved, it appears that the decomposition problems associated with particle penetration can be considerably reduced with the use of such a system.

Although a seal may be effective to the outer wall of the space vehicles, personnel and equipment inside might still be subjected to direct impacts of reduced violence with a particle or to associated penetration effects. This risk can be estimated by a probability analysis.

C. PARTICLE RADIATION

1. General

This section discusses the particle radiation near the earth and in free space, its interaction with material and components of space systems, and the resulting biological effects. It is known from the wealth of literature that the degree of interaction depends on the type of radiation (proton, electron, gamma, etc.), the energy spectrum, the duration of exposure, and finally, the density and geometry of the shielding structure. From space and ground based nuclear radiation experiments, it becomes apparent that particle radiation can result in transient or in permanent damage, depending on both the dose rate and integrated dose.

In order to predict the critical particle radiation that might seriously affect the space system structure and components, space suits, space maintenance capsules and the space workers both inside or outside space systems, the following four areas were investigated:

- (1) The external particle radiation spectra encountered during four selected missions based on the most recent data.
- (2) The internal dose rate and integrated dose; based on the shielding structure and exposure time.
- (3) The effects of particle radiation upon the material used in space systems and upon system components (mechanical, electromechanical and electrical components).
- (4) The biological hazards based on the dose rates and on the biological effectiveness of the impinging radiation.

A study of the space systems that were the object of the contract resulted in a selection of the following four basic missions:

- (1) A circular 300 n.m. (560 km) orbit (4-year duration)
- (2) An equatorial 24-hour orbit at 19,300 n.m. (36,000 km) altitude.
- (3) A lunar trajectory of a manned system, launched from Cape Canaveral by a Saturn booster and injected during ascent.
- (4) A lunar site.

During these missions, space systems will be subjected to a composite particle radiation, originating from different sources, depending on the location in space and on time. Each of these radiation phenomena will be discussed, in terms that permit, in connection with the mission trajectory, a prediction of the external dose rate and integrated dose.

2. The Radiation Environment

There are four important radiation phenomena encountered during the four basic missions of the vehicles launched from Cape Canaveral along the Atlantic Missile Range:

Primary Cosmic (Galactic) Radiation

Van Allen Belts

Solar Flare Radiation

Artificially Injected Radiation

a. Primary Cosmic Radiation Near Earth and in Free Space

The nature of the primary cosmic particles, originating in the solar system or in the galaxy, has been under extensive study for many years. This radiation consists of approximately 86 percent protons, 13 percent alpha particles and 1 percent of light and medium nuclei (Li, Be, B, C, N, O, F). The particles appear to be omnidirectional. The most striking characteristic is the extraordinarily large energy range from 10^8 ev to 10^{18} ev. The average intensity is modulated during the 11 year solar cycle activity. If the solar activity increases, the radiation intensity decreases ("Forbush decrease") and this intensity ratio can be two to one between periods of sunspot minimum and maximum.

At satellite altitudes within the range of a few hundred miles, the cosmic radiation intensity is subjected to the influence of the geomagnetic dipole (39). Meridional profiles indicate an increase from the geomagnetic equatorial plane (minimum) to the polar region (maximum).

The average kinetic energy has been estimated by several authors between 3.6 Bev (40) and 4.0 Bev (39) per nucleon. The overall omnidirectional flux density above the atmosphere as measured by the Pioneer IV satellite was $N = 1.8 \pm 0.3/\text{cm}^2\text{-sec}$. Measurements from balloon-borne equipment corrected to zero atmospheric depth resulted in a lower value of $N = 1.38$. We can therefore, make the conservative assumption that the overall omnidirectional flux will not exceed $N = 2.0/\text{cm}^2\text{-sec}$ and this value and an average kinetic energy of 4 Bev shall be used for the calculations in the following phases.

For lunar missions or lunar surface explorations, the free space flux density as encountered outside the Van Allen belts has to be considered. Again an omnidirectional flux can be assumed (the effect of the very small lunar magnetic field is neglected). Recent measurements of the Pioneer V space probe showed a flux density in free space of $2.5/\text{cm}^2\text{-sec}$. and an ionization rate inside appr. $1\text{g}/\text{cm}^2$ of low atomic number shielding material of 0.6 mr/hr. (39).

This flux density is higher than the values measured by Pioneer IV and high altitude balloons at near earth orbit altitudes. It will be used for the calculation of the composite particle radiation encountered during missions 2 and 3 and for the lunar site mission. The particle distribution is assumed to be identical with that found just above the earth atmosphere.

b. Van Allen Belts

The knowledge of the immense radiation field, temporarily trapped in the geomagnetic field is still incomplete as demonstrated by the most recent results obtained by the Explorer XII Energetic Particle Satellite. This satellite, launched on August 15, 1961 into a highly elliptical orbit (perigee 300 km, apogee 77,250 km) completed 102 orbits in 112 days lifetime and transmitted back to earth more radiation data than all previously launched satellites. The evaluation of the results has not yet been completed. However, the first results obtained from Goddard Space Flight Center ⁽⁴¹⁾ already revealed the important new findings that can be summarized as follows:

- (1) The existence of high energy protons of the order of several ten million ev in the heart of the Inner Van Allen Belt was confirmed at approximately 1.5 earth radii. (Measured from the center of the earth). However, the altitude range of protons extends much larger than previously assumed; i.e. they are trapped to an altitude of at least 8 earth radii. At 3 earth radii, the average proton energy is a fraction of one Mev but their flux density rises to a maximum and equals that of the electrons present ($10^8/\text{cm}^2 - \text{sec.}$). Their average energy ranges from 100 Kev to 400 Kev. The proton flux density now appears to decrease slowly with distance from the earth.
- (2) Also the altitude range of electrons extends farther than previously measured with Explorer I, III, IV, and Pioneer III and confirmed by Sputnik III and Mechta. Soft electrons with energies of several ten-thousand ev were found from 6 earth radii to the outer edge of the magnetosphere. (The outer edge varies daily from 8 to 12 earth radii).
- (3) The flux density of electrons in the heart of the outer Van Allen Belt is about 1000 times lower than the previous estimate of $10^{11}/\text{cm}^2 - \text{sec.}$ In other words the highest flux density of electrons with energies above 40 Kev does not exceed $10^8/\text{cm}^2 - \text{sec.}$ Figure 29 presents a summary of these data. This result confirms an assumption made by Dessler ⁽⁴²⁾
- (4) The outer edge of the trapped particle region exhibits an abrupt discontinuity; the low energy electron radiation falls to the free space radiation described in the foregoing section.

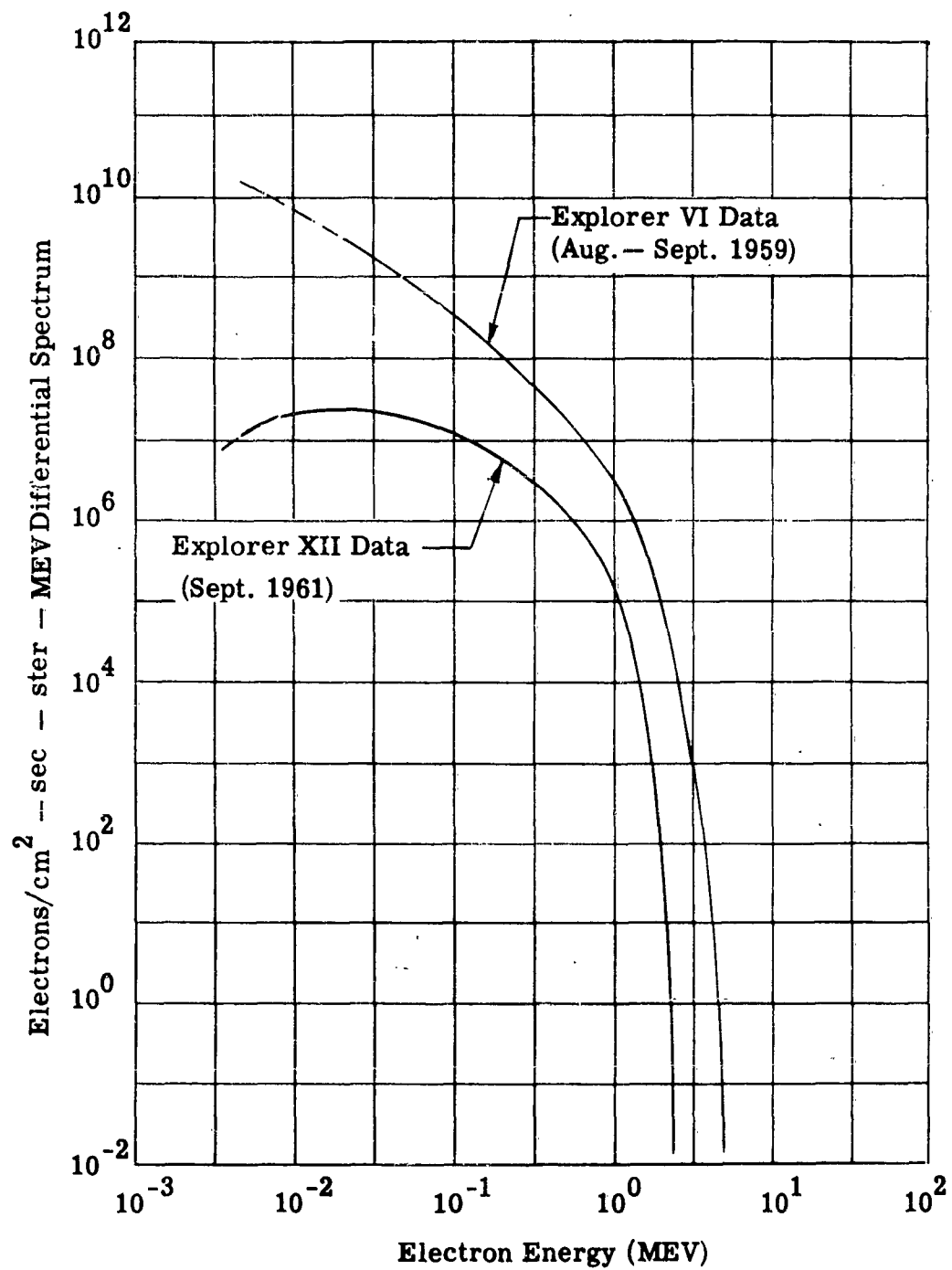


Figure 29. Electron Distribution in Outer Van Allen Belt

These results shall be used to estimate an intensity distribution of the low energy protons, ($120 \text{ Kev} \leq E_p \leq 4.5 \text{ Mev}$), i.e., the intensity in the geomagnetic plane versus the distance from the earth's center. Also the intensity distribution of electrons with energies $E_e \geq 40 \text{ Kev}$ is estimated. The distribution distance function of Figure 30 demonstrates the slow decrease of the intensity of both particles. Of course, there is still a large uncertainty. As more data from Explorer XII become available, a correction of the distribution curves will be required. Included in Figure 30 is the well known intensity distribution of high energy protons and electrons found in the Inner Van Allen Belt.

Based on the electron intensity gradient of Figure 30, an estimate was made of the spatial distribution under the assumption of an isointensity field pattern similar to that used in Reference 42. The resulting distributions are shown in Figures 31 and 32.

During two of the missions studied during this contract, (Circular 300 n.m. orbit and lunar site), the vehicle will be subjected to only primary cosmic particles; however, a possibility of radiation by high energy solar flare particles exists during all four missions. The solar flare radiation is discussed in a later section. During passage through the Van Allen Belts, the vehicle and crew are subjected to the intense proton and electron fluxes previously described. Intensity rate and integrated intensity depend on the mission trajectories with respect to the belts and the time spent in each of the intensity regions. For an estimate of these two radiation quantities, ascent trajectories for a 24-hour orbit and a lunar mission were calculated as a function of time. From these data, mission coordinates relative to the geomagnetic plane were determined. For both missions, the injection parameters were selected compatible with the capability of a Saturn booster, launched from Cape Canaveral along the Atlantic Missile Range. The injection occurs somewhere over the Atlantic Ocean. The trajectories were computed to a distance of 60,000 km from the center of the earth, and the results are presented in terms of geomagnetic latitude and radial distance from the earth's center in Figures 31 and 32.

Some remarks with respect to the 24-hour orbit are in order. The satellite is assumed to be launched into a 100 n.m. parking orbit with an inclination of 30.4° . It coasts in this orbit to a position north of the first equatorial crossing, where it is accelerated to flight conditions required to achieve an elliptic orbit whose apogee is at 36,000 km (24-hour orbit altitude) and is in the equatorial plane. When the coasting satellite reaches its apogee, it is again accelerated to the required circular orbit velocity. Also the inclination is changed to 0° to obtain an orbit in the equatorial plane. Then the satellite will remain stationary in an earth geographic coordinate system and also in the geomagnetic system. The geographic coordinates of the final position are 0° latitude and 101.2° longitude; the corresponding geomagnetic coordinates are 11° S latitude and 171.1° E longitude.

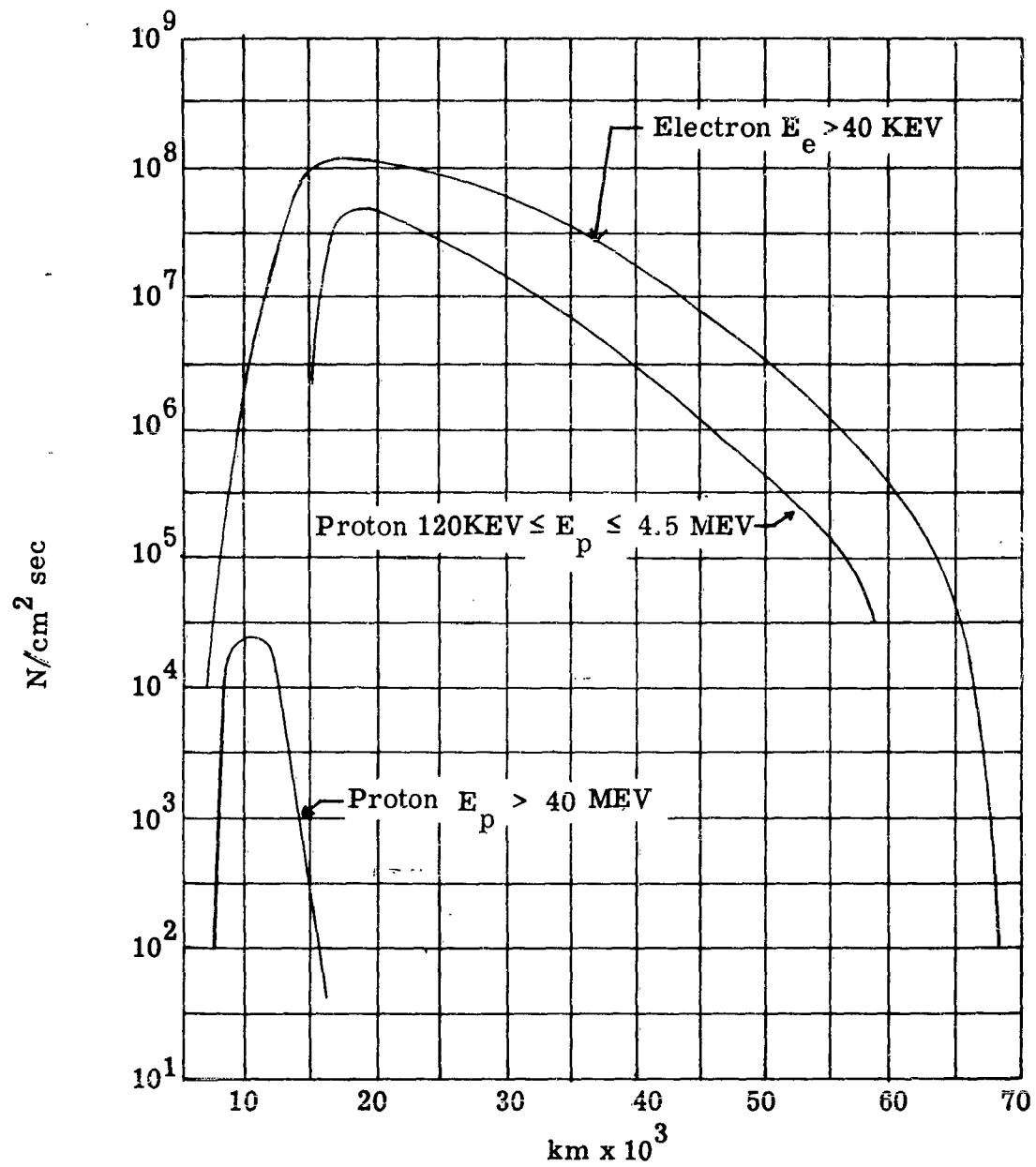


Figure 30. Estimated Intensity of Trapped Protons and Electrons

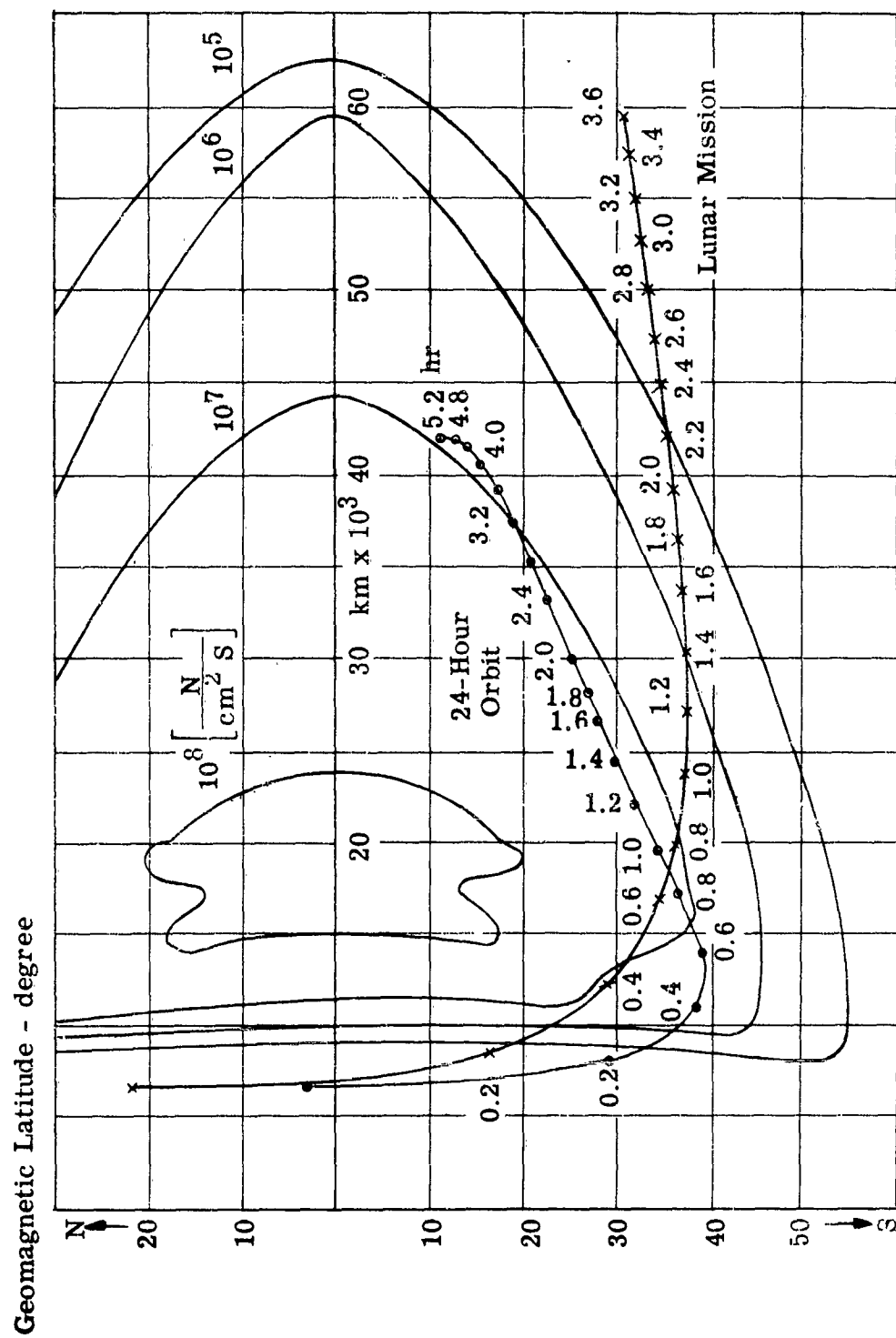


Figure 31. Isointensity Lines for Electrons ($E_e > 40$ KEV)

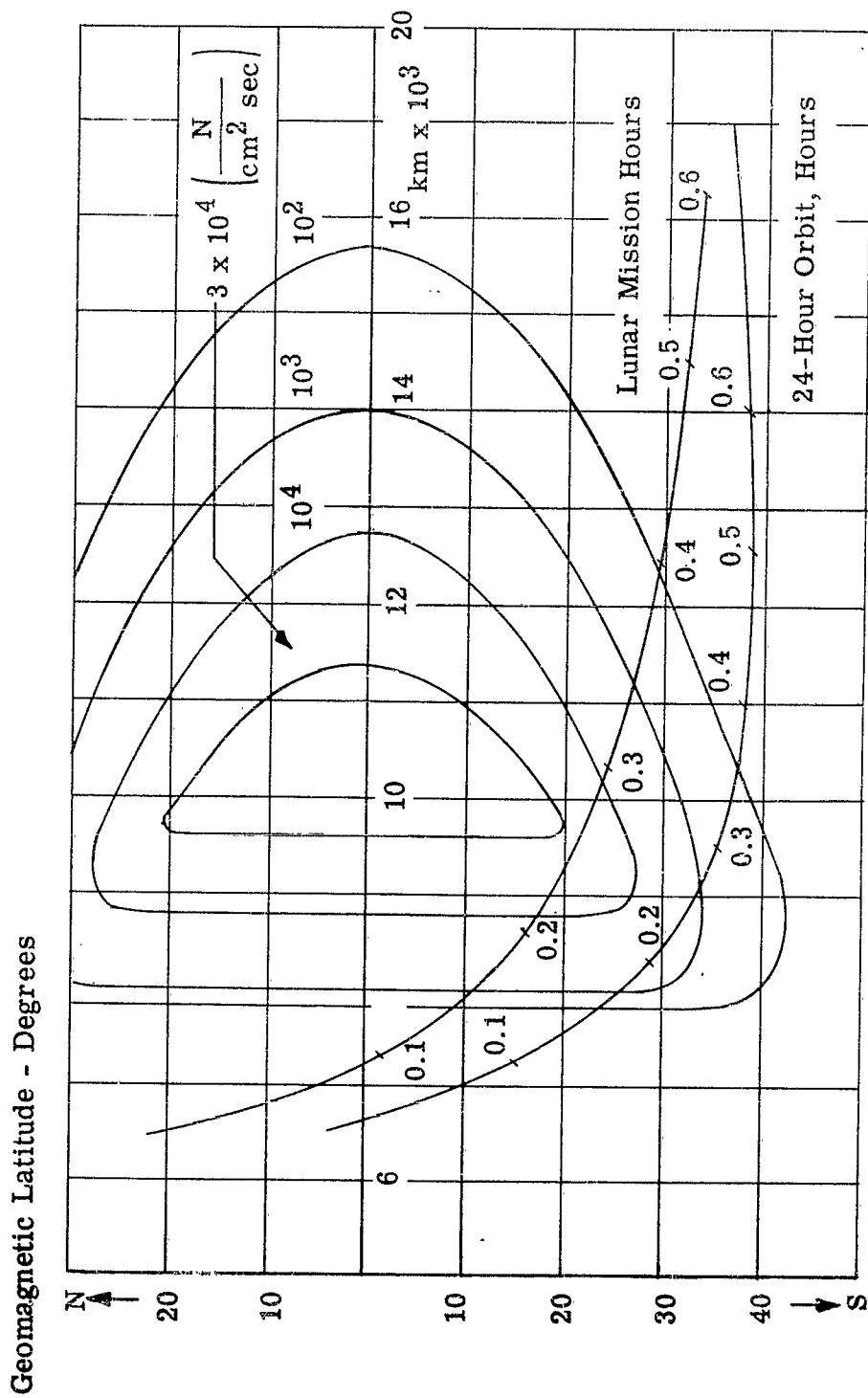


Figure 32. Isointensity Lines for Protons ($E_p > 40 \text{ MEV}$)

Integration of the intensity rates over the flight time inside the Van Allen Belts results in the intensity-time function for electrons and protons, shown in Figures 33 and 34 for the lunar mission and in the Figures 35 and 36 for the 24-hour orbit. The flight time through the proton (inner) belt during the lunar mission is only 0.4 hour and through the electron (outer) belt is 2.4 hours.

The 24-hour orbit requires one passage through the proton belt; the satellite then remains at an altitude of 36,000 km in a region where the electron intensity is approximately $10^7/\text{cm}^2 - \text{sec}$. Consequently, the integrated electron intensity increases with each orbit made as can be seen from Figure 35.

c. Solar Flare Radiation

The phenomena associated with high energy protons, ejected from solar flares, are complex and their interpretation is difficult because of the low number of events investigated during the period from the first detected relativistic energy event in 1942 to the present time. It is not intended in this section to describe solar flare phenomena in detail as several papers recently published present extensive reviews on solar flare radiation (39, 44). However, a short summary of the principal characteristics shall be presented:

Two groups of solar flare events exist. The relativistic events with energies high enough to be detected by sea level instrumentation occur about once every four years on the average. Since their first detection in 1942, ten additional events were measured at sea level; five indicating energies exceeding 15 bev. The maximum intensity of the February 23, 1956 event was estimated at 10^5 particles/ $\text{cm}^2 - \text{sec}$. The intensity I of relativistic events decays rapidly near the earth according to the equation:

$$I = I_0 \frac{t_0^2}{t}$$

I_0 = initial intensity at time t_0 about one hour from start of the flare.

No obvious correlation between the occurrence of relativistic events with the solar cycle has been found.

During the International Geophysical Year, a second group of solar flare radiation events was discovered by satellites, balloons, and ionospheric techniques, which are called low energy or nonrelativistic events with energies between 10 to 500 mev. They occur about once per month during periods of high solar activity. The most recent Explorer XII experiments indicated that the frequency of these events might be higher.

The low energy protons arrive in the vicinity of the earth approximately one hour after the beginning of a major flare of medium intensity but generally of lower intensity and they are observed over the polar cap (north of 60° geomagnetic latitude). Their intensity may grow during several hours, then decay according to the same intensity law found for relativistic events. Also a sudden increase of intensity during decay can occur as measured by Explorer XII on September 30, 1961.

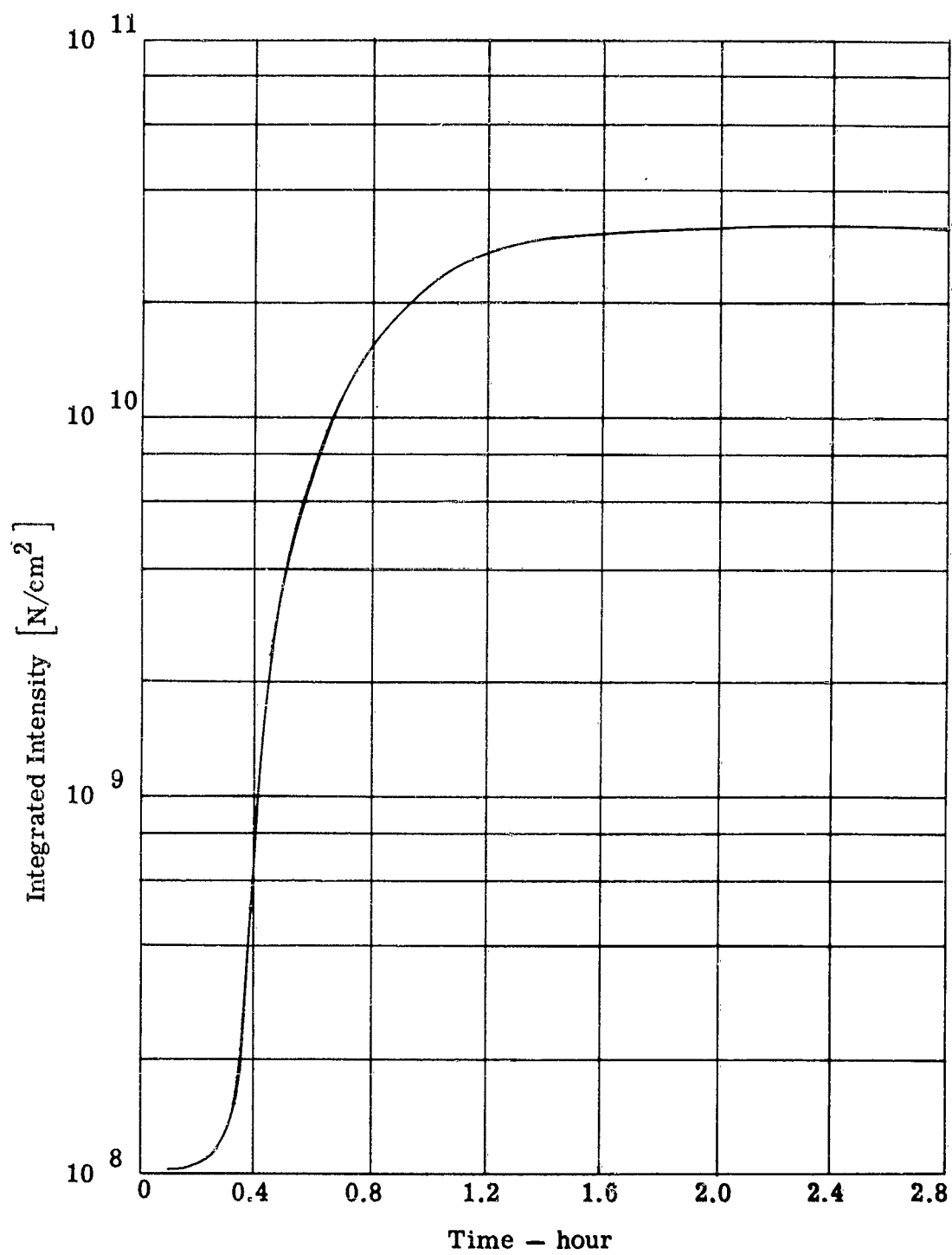


Figure 33. Integrated Intensity of Electrons for a Lunar Mission
(Outer Belt)

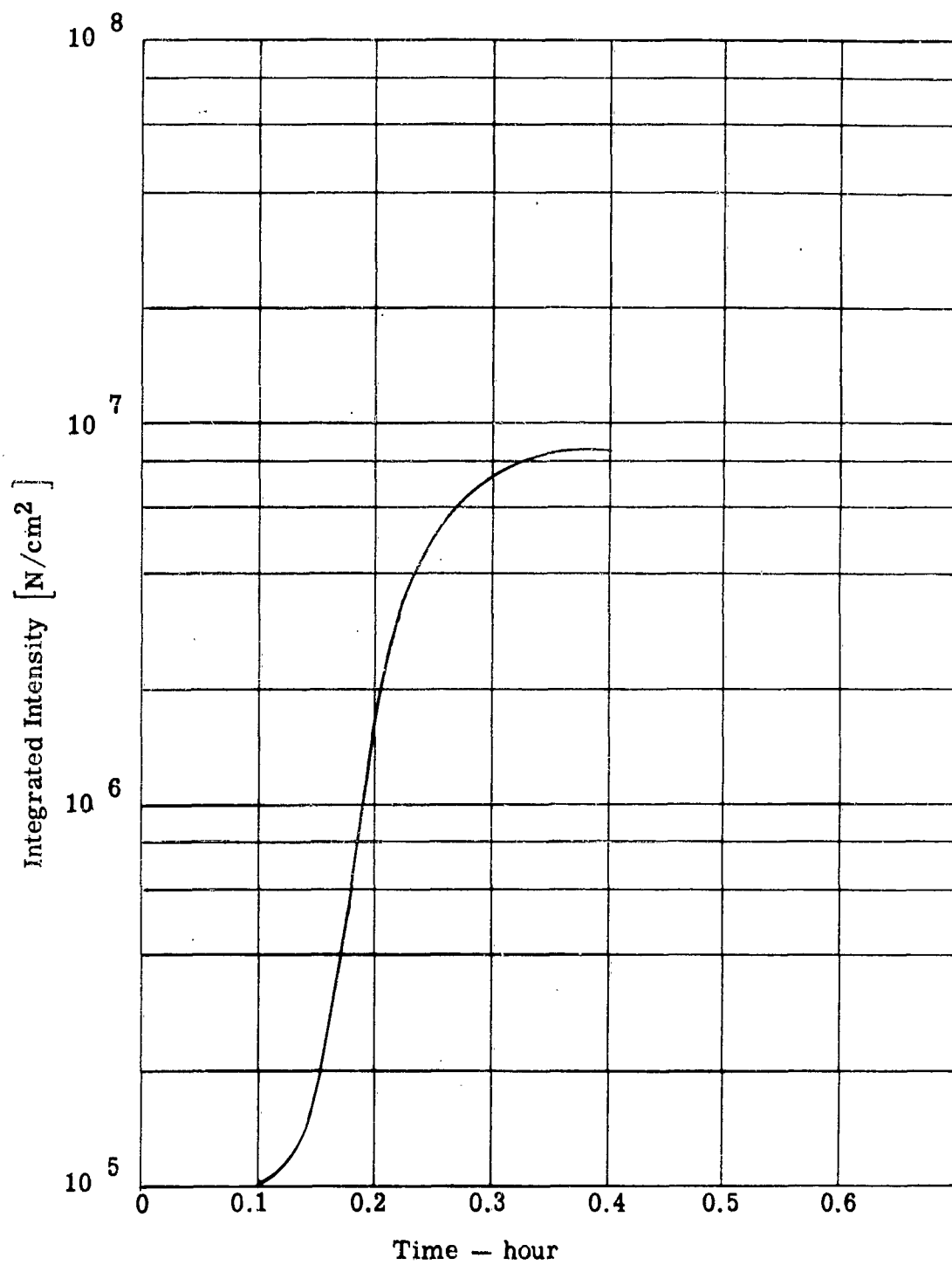


Figure 34. Integrated Intensity of Protons for a Lunar Mission
(Inner Belt)

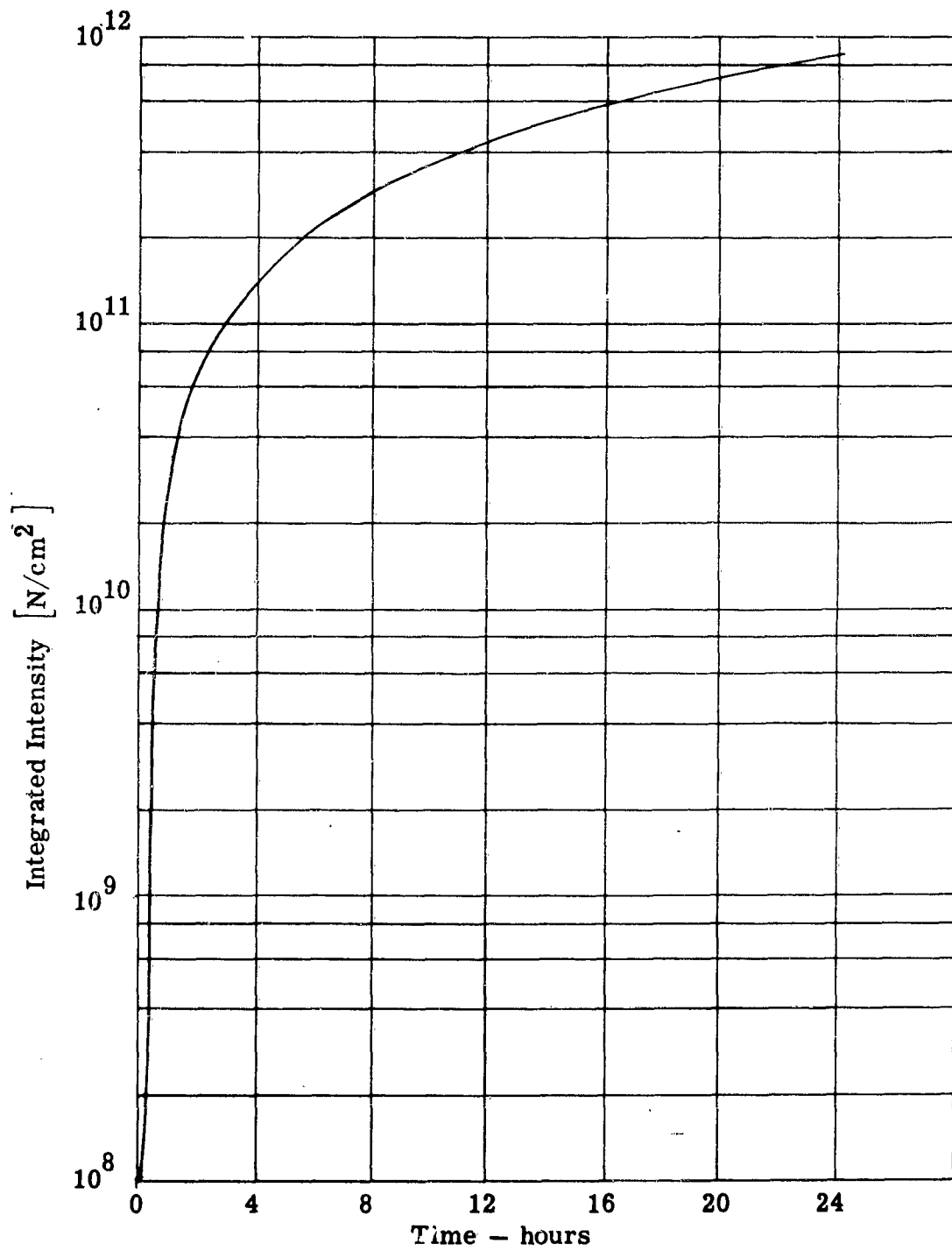


Figure 35. Integrated Intensity of Electrons for a 24-Hour Equatorial Orbit (Outer Belt)

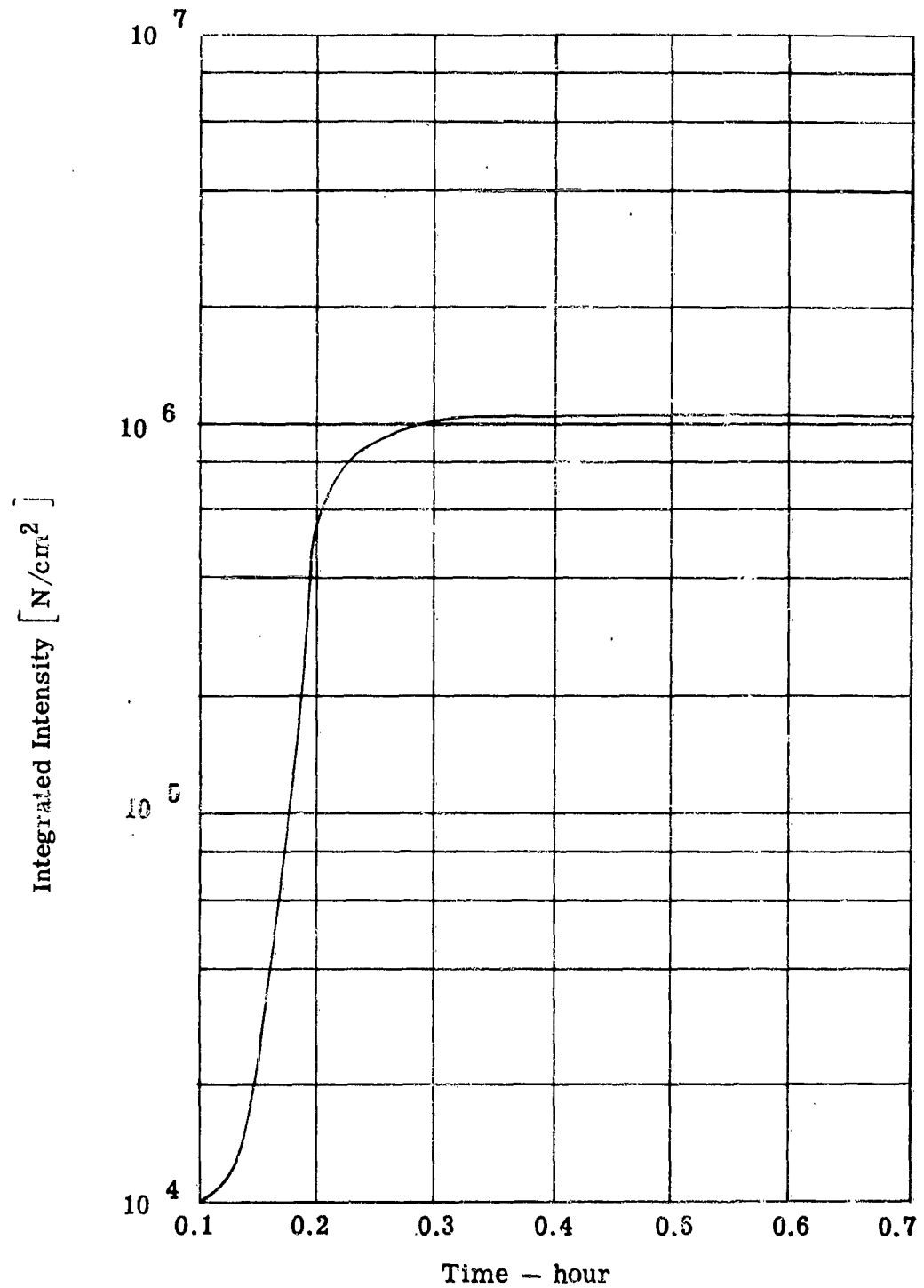


Figure 36. Integrated Intensity of Protons for a 24-Hour Equatorial Orbit (Inner Belt)

The intensity-time function for both high and low energy solar flares is demonstrated in Figure 37 derived from Schaeffer (49). High energy flares completely degrade in approximately 50 hours; the low energy flares need periods longer than 100 hours to completely disappear. It can also be seen from this figure, that the peak intensity can vary over a large range of many decades from values just detectable above background radiation to an intensity of 10^3 protons/cm²-sec-ster., and more.

An evaluation of the sunspot activity, observed during the last 100 years has indicated an eleven year variation. The number of occurrences of nonrelativistic events is related to this cycle. The number reaches a maximum approximately one or two years later. The present sunspot cycle began in 1954, reached a maximum late in 1957, and will probably have its minimum in 1965. Consequently, the number of solar flare events is expected to decrease considerably during the period of minimum sunspot activity in 1965 and will again reach a peak in 1968 or 1969.

d. The New Artificially Injected Radiation Zone

On July 9, 1962, the United States detonated a nuclear weapon of a 2 megaton yield at 200-mile altitude creating a new zone of radiation. The lower altitude limit of the new belt is ragged. It begins at about 200 miles over the mid-latitude in the southern hemisphere, and at an altitude of about 500 miles over the equator. The upper limit is believed to extend to an altitude of 800 miles.

NASA has stated that the intensity of electrons to be encountered in the lower part of the new belt should be too low to constitute a hazard to an astronaut. However, the zone has already caused deterioration of the solar cells in the Navy's TRANSIT IV-B and TRACE satellites which are presently in Earth orbit.

This experiment and those conducted during Project ARGUS in mid-1958, have demonstrated that a high radiation intensity of trapped, high energy electrons can be produced artificially in selected regions of earth-space for short periods of time. A theoretical upper limit to the stored intensity of radiation in such a quasi-stationary state may be estimated by taking the total volume density of kinetic energy of charged particles equal to the earth's magnetic density $B^2/8$ at the point in question. However, this result is not particularly illuminating from an engineering standpoint unless the particle kinetic energies are known.

For the purposes of this present study, a detailed consideration of such a transient radiation environment was not warranted. However, it is obvious that the phenomenon should not be ignored when attempting to maximize the probability of system malfunction or when defining the total interpreted dose received by a space maintenance worker.

Protons with
Minimum Energy

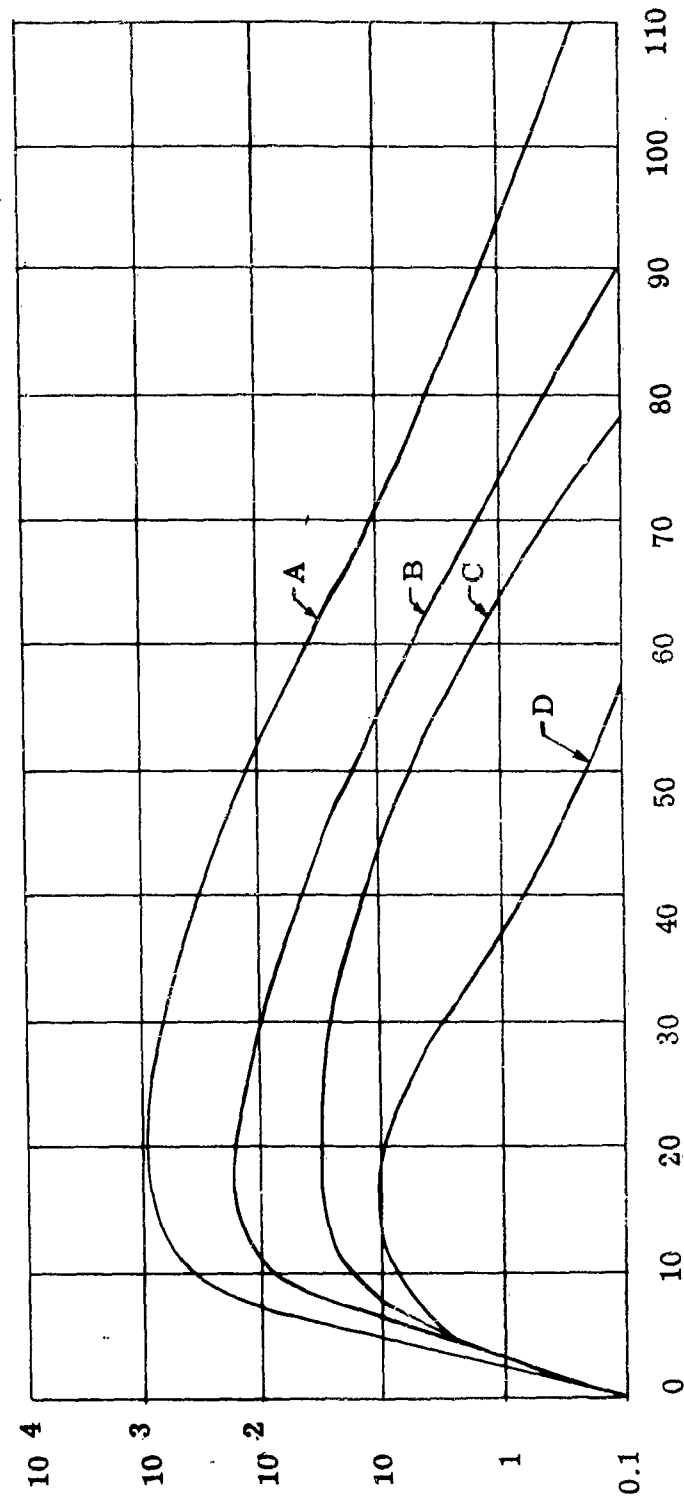
A = 43 MEV

B = 94 MEV

C = 140 MEV

D = 209 MEV

Proton/cm² - sec-sterad



Time After Onset of Solar Flare - Hours

Figure 37. Solar Flare Decay with Time

3. Internal Radiation Levels

This section examines the anticipated dosage due to the above described radiation environments as measured under the wall materials of the vehicles of interest and an extravehicular, full-pressure suit.

Since the external radiation environments are not known with a high degree of accuracy, the estimates of internal doses are at best order-of-magnitude calculations. However, for consistency, and to achieve a degree of conservatism, maximum values were calculated. It is felt that this approach is warranted considering the fact that other possible environments, combined with particle radiation, may produce unexpected limitations.

For the following calculations, it is assumed that the structure of all vehicles has a surface density of approximately 1.5 g/cm^2 . The vehicle structures were assumed to consist of a laminated wall with a low Z material on the outside and a denser material (high Z) on the inside. The space suit was assumed constructed in a similar fashion with a smaller thickness of high Z material.

The inner Van Allen belt should not affect any system which operates at altitudes below 900 km. However, for the 800 km orbit the radiation which might be encountered is that due to the "horns" of the outer electron belt which extend to lower altitudes at latitudes between 40° and 60° . At latitudes between 0° and 40° and between 60° and 90° , there is a negligible contribution to the radiation dose due to the "horns."

The systems for use in the 560 km orbit will be subjected to outer Van Allen belt horn radiation with superimposed galactic and solar flare radiation. The systems for the other 3 missions will be subjected to all the types of radiation and particles because they include traversal of both Van Allen belts, and bombardment by galactic rays and solar flares. The additional dose due to a lunar site environment has been estimated separately.

Also considered to a limited degree was the secondary radiation, i.e., Bremsstrahlung, from electrons and secondary gammas from proton-bombardment. Bremsstrahlung has been estimated from an electron spectrum for an 800 km altitude (43-50) from Explorer VI space probe data. Since the electron differential spectrum used to that for high latitudes and altitudes from 800 to 1000 km, the estimate is assumed to be applicable to a 560 km orbit. The Bremsstrahlung has been estimated from the electron spectrum without correction for temporal or galactic ray-solar flare interactions.

The proton spectra for the inner belt and for solar flares. Also the spectra for galactic cosmic radiation were obtained from References 46-48. The secondary resultant low dose radiation dose from protons was considered. However, due to the low dose rate (Figure 38), this source rates obtained with shields of low surface density (1.5 gm/cm^2) of radiation dose is insignificant and is not considered further.

Differential Spectrum

$$n = 3.1 \times 10^9 E^{-4.6} (E > 85 \text{ MEV})$$

$$n = 1.0 \times 10^6 E^{-2.8} (E > 85 \text{ MEV})$$

$$n = \frac{P}{\text{cm}^2 \times \text{sec.} \times \text{Str.} \times \text{MEV}}$$

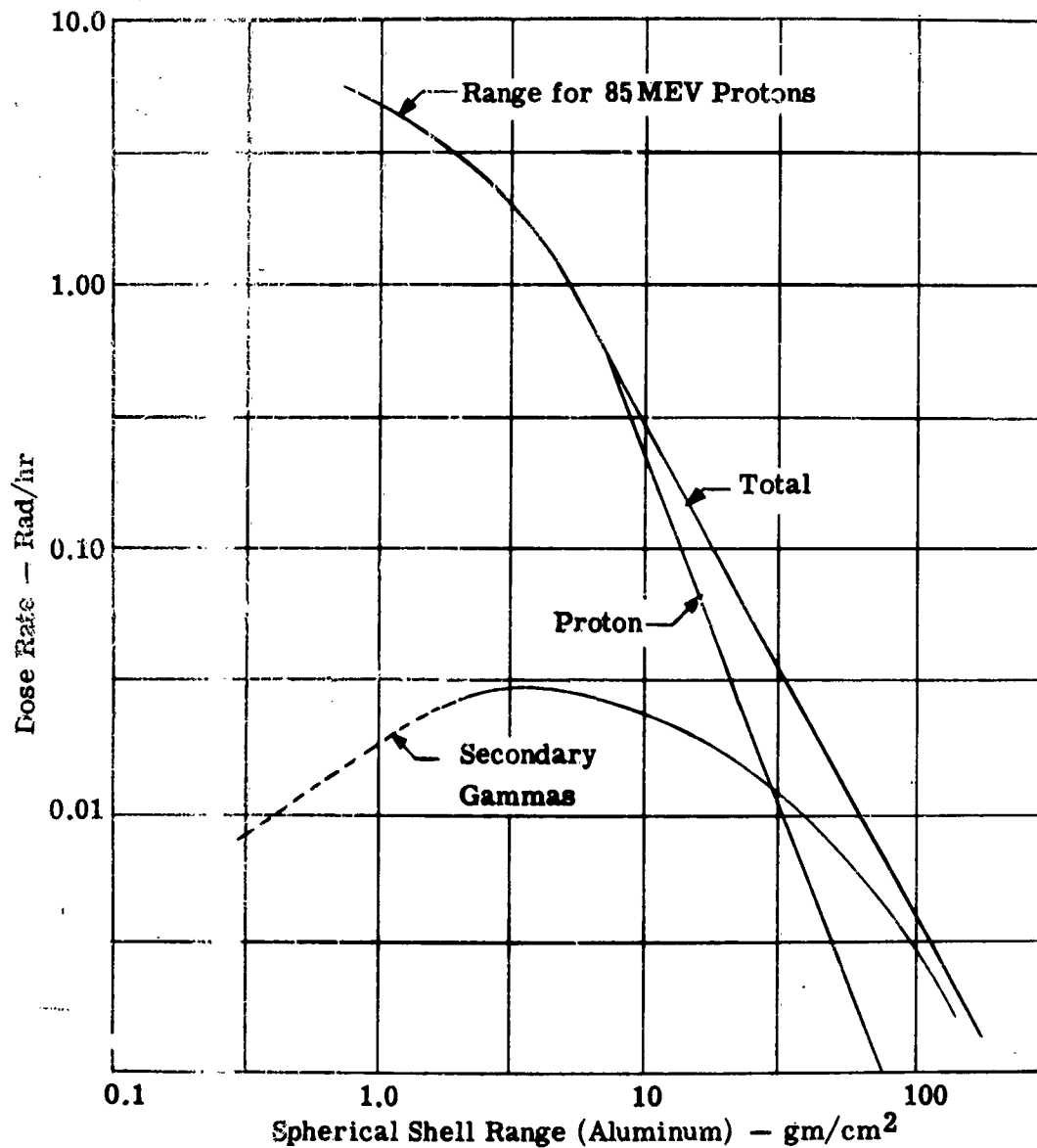


Figure 38. Gamma Dose for a Solar Flare

The principal sources of radiation for the lunar site mission will be unattenuated solar flares and galactic cosmic rays because of the absence of an atmosphere.

a. Internal Dose Rate Calculation Methods

The radiation flux levels and energies which will be encountered in space travel above 800 km altitude are a complex function of:

- (1) Time - determining the presence or absence of solar flare or storms and if the outer Van Allen belt is extended or contracted due to solar storms.
- (2) Position - determining the number and energy of the radiation impinging on the vehicle due to the geomagnetic forces at the particular position.

Table XIX describes the radiation sources which will determine the radiation dose during each mission.

TABLE XIX
RADIATION SOURCES FOR VARIOUS SPACE MISSIONS

Mission	Van Allen Belts	Cosmic Rays	Solar Flares
560 km orbit	outer only	attenuated	attenuated; time dependent
traversal for 24-hour (36,000 km) orbit	both inner and outer	variable attenuation	variable attenuation; time dependent
24-hour (36,000) orbit	outer only	unattenuated	unattenuated; time dependent
traversal for lunar mission	both inner and outer	variable attenuation	variable attenuation; time dependent
lunar site	none	unattenuated	unattenuated; time dependent

The principal cause of attenuation is the interaction between the solar flare and galactic cosmic rays with the earth's magnetic field and its atmosphere. As discussed previously, the time dependence is due to the 11 year solar flare cycle. A slight time dependence is also exhibited by galactic cosmic radiation, but because of their extremely high energy, the effect on absorbed radiation dose is slight. Thus, any time dependence of galactic rays is ignored in the following calculations.

The extent of the Van Allen zones has exhibited some temporal dependence. The inner proton belt has varied slightly over the period of observation and therefore the effect of this variation was not considered. However, the outer belt is primarily composed of electrons, and varies extensively due to solar and galactic cosmic ray impingement. Because of the complex interaction of these events, the environment used in the calculations was that corresponding to the most hazardous time. Proton variation in the outer belt also varies due to injection of solar flare protons. These resulting proton spectra are shown in Figure 39. The most energetic electron spectrum was obtained from Explorer XII space probe as described before. Therefore, using a spectrum developed from Explorer XII data, although it is not fully analyzed at present, will give a better estimate of the maximum amount of radiation damage to be encountered in space.

The data presented by References 43 - 48 were used in order to develop an accurate estimate of the composition of the Van Allen belt regions and deep space radiation levels. As a result, the following two quantities were calculated: (1) An "average" energy spectrum for these radiation belts under the assumption that the time dependent changes in the spectrum can be omitted; and (2) an estimate of the particle number flux, to which a vehicle would be subjected at particular points along its trajectory or orbit.

The concept in formulating this number-energy spectrum was based on a mass-energy balance around the vehicle in a quick earth-escape trajectory for each point in space. By utilizing the presently known information on the geomagnetically trapped radiation, the geometry of representative space vehicles and the space suit and the material of the vehicle walls or suit, an estimate of the absorbed dose by the vehicle and/or suit was made.

For this mass-energy balance, a model has been described^(44, 47) which relates the spatial distribution of protons in space to an earth-centered space coordinate system. The general result derived from this model is that the closest (to the earth) section of the inner Van Allen belt contains a relatively large number of high energy protons. At higher altitudes the energy of the protons is low; and they are fairly evenly distributed at different altitudes. The most widely accepted generating mechanism for these spectra is the decay of albedo neutrons, which provide the high energy protons near the point of greatest resistance, the Earth's atmosphere. Because the resistance of the outer Van Allen belt is low, the protons from the decay of the free albedo neutrons migrate to the inner belt. Noting that the energy spectrum can be approximated by a Gaussian-distribution, the spectra for each of the zones of interest are shown in Figure 40.

The description of the electron spectrum of the outer belt required lumping the known maxima together and assuming an energy spectrum which was also Gaussian in shape. Based on this assumption, the resulting energy spectrum can be represented as a straight line on normal probability charts. The electron probability function is presented in Figure 41. Finally, by assuming that lines of

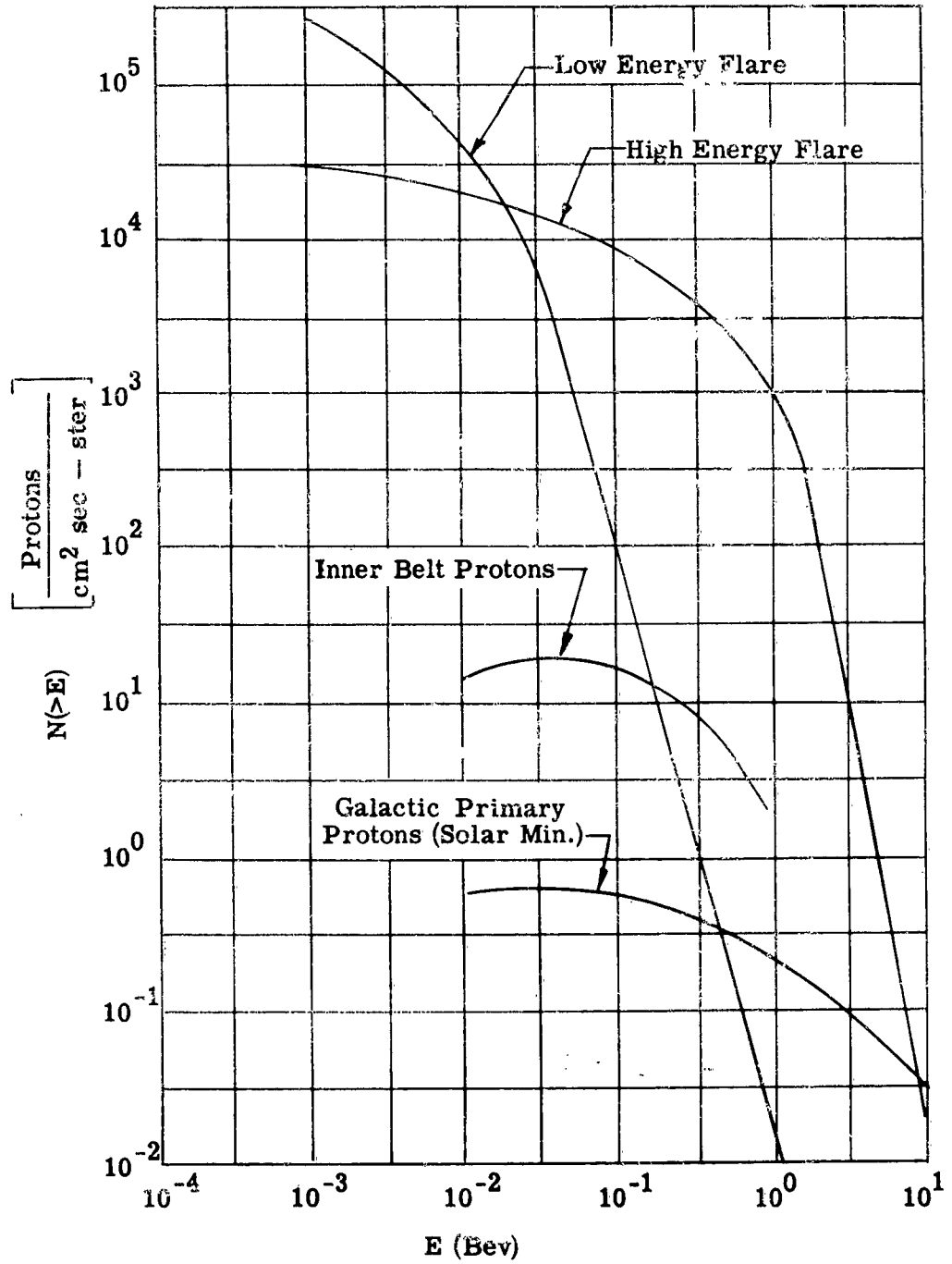


Figure 39. Integral Energy Spectra for Solar Flares, Inner Belt Protons and Galactic Primary Protons

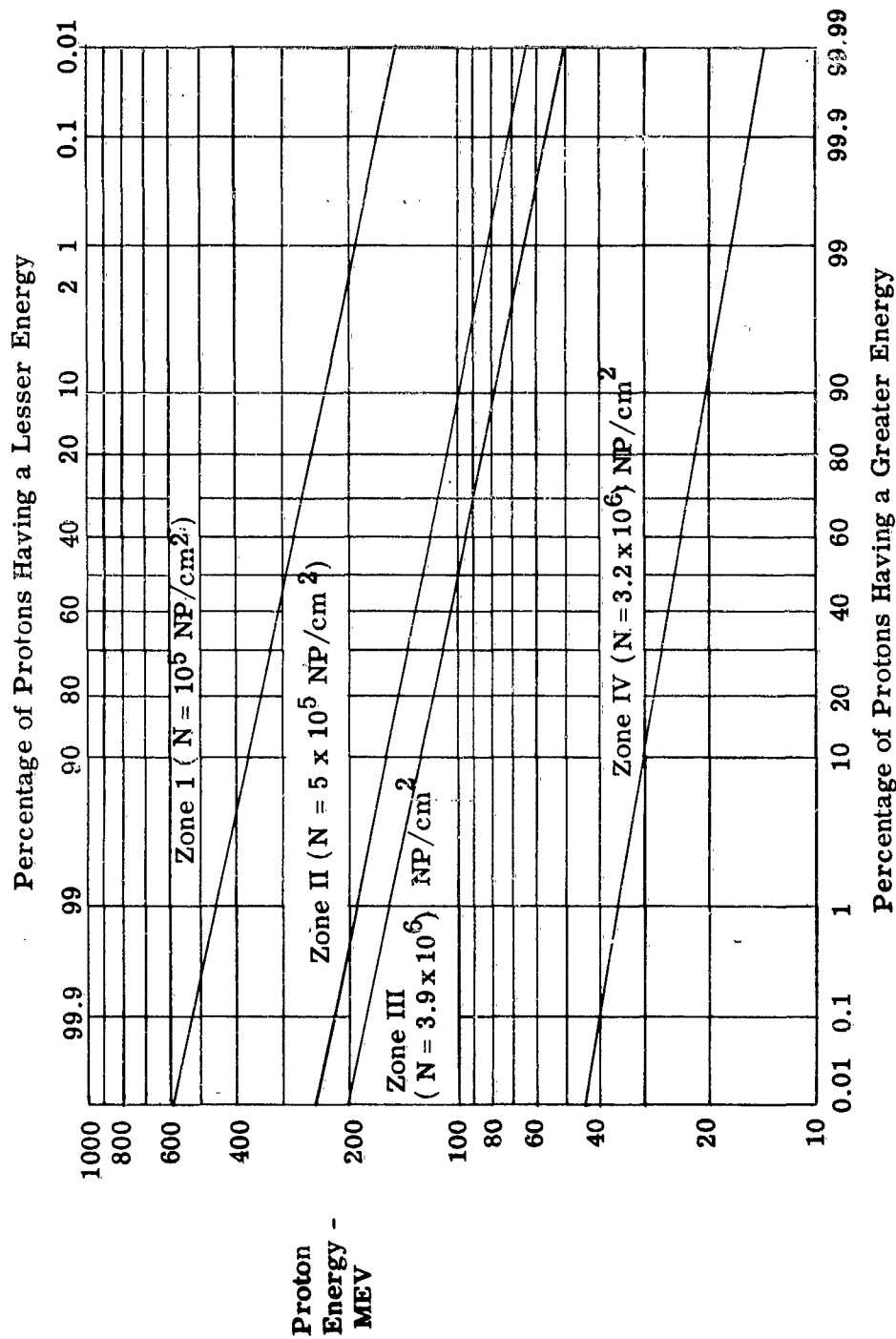


Figure 40. Proton Energy Spectrum Impacting 1 cm² of Vehicle (Lunar Trajectory)

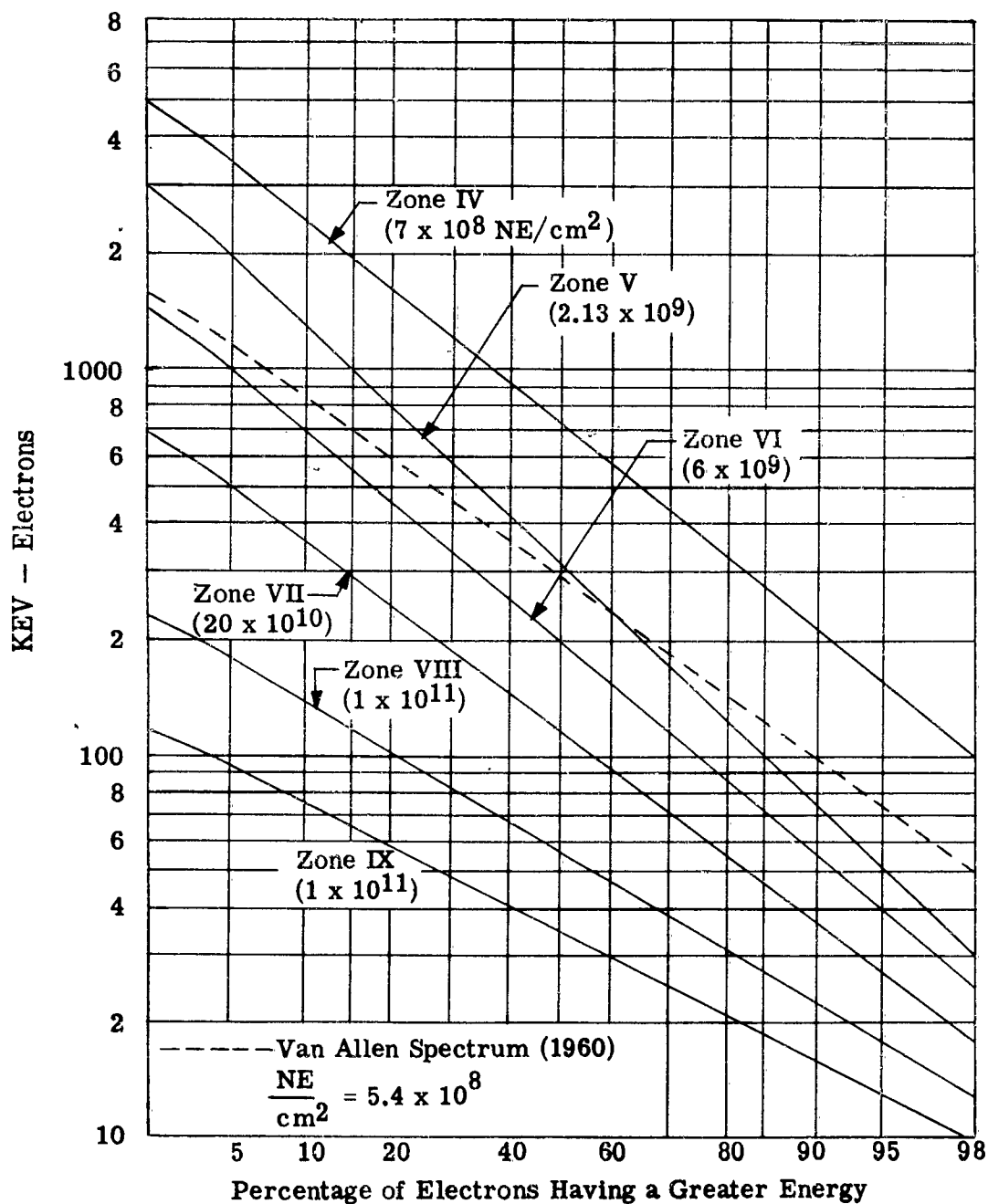


Figure 41. Electron Energy Spectrum Impacting 1 cm² of Vehicle (Lunar Trajectory)

constant geomagnetic intensity are lines of constant particle energy, it was possible to calculate spectra for each zone of the space vehicle trajectory.

b. Proton Calculations

The procedure applied in proton calculations is described in Reference 48 which may be consulted for details of the method. The wall surface density for the vehicle is 1.5 gm/cm^2 and for the space suit is 0.08 gm/cm^2 . The wall material for the calculations was assumed to be aluminum and no estimate of radiation dose variation due to combination of several wall materials was made.

Calculations were performed using steady, time-average, proton spectra for the inner Van Allen belt and for the galactic rays. Time variations were accounted for in the proton dose calculation for solar flares. In order to provide a method, which eliminates time dependence until the last calculation, all possible proton energies in the spectrum were grouped and the effect of each energy group on the above walls was evaluated. After the radiation dose for each proton energy group was obtained, any time effects were applied prior to integration to obtain the total dose.

The proton spectra for the inner Van Allen belt and galactic rays are described in References 44-47. The number of protons of each energy group range was obtained from equations relating magnetic field strength and quantity of protons with the latitude and longitude along the vehicles trajectory or orbit. Time dependence was neglected for both of these spectra as there are small variations in energy and number of protons, consequently resulting in a small contribution to the total dose.

In the same way the solar flare proton dose was calculated and corrected (for the time dependence) to that occurring at the height of the solar flare maximum. A typical time dose rate function behind walls of two different surface densities with a 60 hour-high energy solar flare is shown in Figure 42.

The calculations indicate the following conclusions:

- (1) Below 40 MEV, protons will not penetrate into a vehicle wall and below 6 MEV into the space maintenance suit.
- (2) Protons with energies between 40 and 150 MEV will be major contributors to the radiation dose, because they do not contain enough energy to penetrate both vehicle walls and the air space or space vehicle occupant. Their absorption coefficient is also very large because of their low kinetic energy. Calculations indicate a dose of 0.6 rads/hour for representative space vehicles, and 1.0 rads/hour for the space suit.

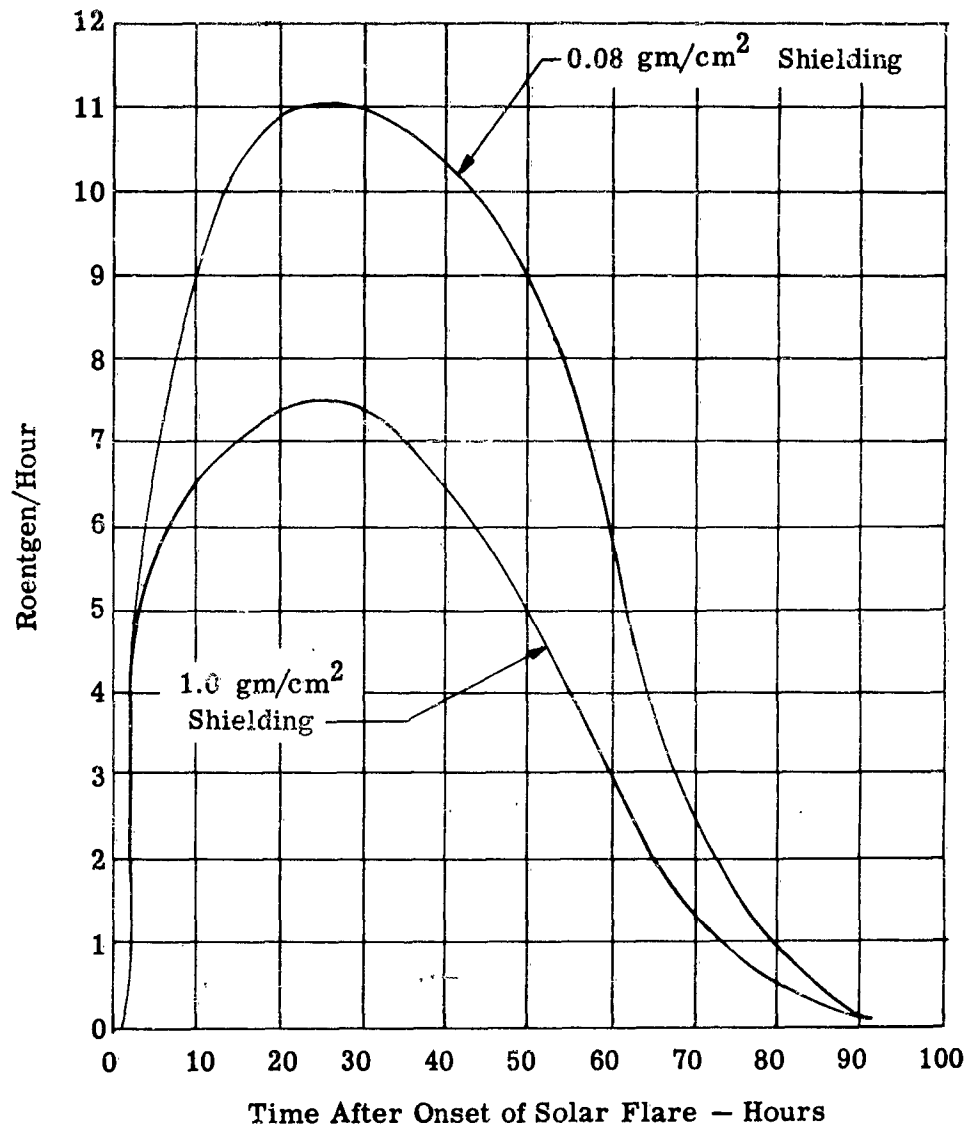


Figure 42. Dose Rate Behind Shielding for Solar Flare versus Time During Flare

- (3) Protons with energies between 150 and 260 MEV will be the next highest contributor to the radiation dose, but because of their lowest absorption coefficient, the dose will be significantly lower than that due to energies between 40 and 150 MEV. This source of radiation will deposit about 0.1 rad/hr behind a 1.5 gm/cm^2 wall, and approximately 0.8 rads/hour to a space suit occupant.
- (4) Protons with energies greater than 280 MEV will not result in a great amount of absorbed radiation as they will penetrate the walls and the air space or occupants in such a fashion as to deposit only a small amount of radiation in these objects.
- (5) The above calculations are for radiation doses caused by protons directly in the objects of interest. However, it has been noted that the protons above an energy threshold of 50 MEV can cause (p, n) reactions forming spallation products. These secondary products may cause a total dose equal to or greater than the primary dose. The spallation products or the secondary neutrons can interact with the shield and produce gamma radiation which is shown in Figure 38 for a given shield. This radiation source (i.e., secondary spallation products) should be evaluated in detail for each particular vehicle wall construction.

In the same way as discussed above, the internal dose rate from solar flare protons was calculated and corrected for time dependence to the height of the solar flare cycle. A typical time function of the dose rate behind walls of two different surface densities for a high energy flare with a duration of 60 hours is shown in Figure 42. A typical total dose behind a $1.5/\text{cm}^2$ wall would be 200 rads per event, and would be increased behind a 0.08 gm/cm^2 wall (space suit) to 440 rads.

Additional calculations have been made to determine the dose rate for a less intense solar flare impinging on a vehicle. Since a less intense flare requires a longer dissipation time, a higher radiation dose for a space suit occupant might result. The vehicle wall would minimize any significant changes in dose to its occupants, because the vehicle wall would remove these lower energy protons. Of course, an astronaut in a space suit would not be continuously subjected to solar flare radiation though this possibility does exist. We may assume, for such low intensity flares, that the dose rate would be equivalent to that for the Van Allen belt protons behind the same amount of shielding (0.08 gm/cm^2).

Calculated integrated dose rate values range from 2-4 rads/hour with a total dose of 180 to 250 rads for a 50 to 60 hour period.

A summary of solar flare doses, calculated for an essentially unshielded vehicle having an aluminum structure (1.5 gm/cm^2), is presented in Table XX.

TABLE XX
DOSES FOR VARIOUS SOLAR FLARES

Radiation Type	Initial Average Energy	Behind 1.5 gm/cm ² Aluminum
Relativistic Solar Flare	400 MEV	5×10^1 rad
High Energy High Flux Solar Flare	50 MEV	3×10^4 rad
High Energy High Flux Solar Flare	40 MEV	1×10 rad
High Energy Low Flux Solar Flare	40 MEV	1×10^{-2} rad

It can be seen from this table that relativistic solar flares of high energy cause smaller doses than the less energetic flares. This difference can be explained by considering the proton energy loss per distance travel in an object. If the solar flare protons are highly energetic, they will penetrate an object without depositing much energy.

The absorbed proton dose rates (R/hour) behind wall surface densities of 0.08 and 1.5 gm/cm² for different missions is given in Table XXI.

The Van Allen belts will cause the highest constant dose rates behind the two shields considered with galactic rays being second highest. Secondary gamma doses from proton bombardment were not considered due to the low surface density of the shields considered.

Solar flare was the only sporadic source of radiation considered.

c. Electron Dose Rate Calculations

The calculation procedure employed is described in detail in Reference 48. Electron bombardment gives rise to two effects: the Bremsstrahlung and the direct electron deposition to an internal component or occupant. The electron spectra employed in these calculations are for the outer Van Allen region (Explorer XII) shown in Figure 43 along with the electron spectra from the Explorer VI probe.

These spectra were considered most important because:

- (1) The known inner Van Allen belt electron spectra are composed primarily of low energy electrons, which are easily absorbed in the shields considered. Therefore, they would not significantly affect the absorbed dose for objects behind the shield.

TABLE XXI
PROTON DOSE RATES (ROENTGEN/HOUR)

Mission	Radiation Source					
	Van Allen Belt		Galactic Cosmic Rays		Solar Flare	
	Suit	Vehicle	Suit	Vehicle	Suit	Vehicle
550 Km Orbit	9×10^{-3}	5×10^{-3}	1×10^{-3}	1×10^{-3}	3.2	2.0
36,000 km Orbit	1.9	1.2	3×10^{-3}	2×10^{-3}	5.3	3.3
Lunar Site	0.	0	1.3×10^{-2}	8×10^{-3}	4.3	2.7*
*The Moon faces opposite the sun during full moon on earth.						
Traversal for						
550 km Orbit	-	5×10^{-3}	-	1×10^{-3}	-	2.0
36,000 km Orbit	-	1.1	-	2×10^{-3}	-	3.3
Lunar Trajectory	-	1.1	-	8×10^{-3}	-	2.7*

- (2) All other sources of electrons from proton-spallation products due to solar flare or galactic rays are considered small. The electrons from the albedo neutrons are considered part of the Van Allen belt environment.

In comparing these spectra, it will be noted that the Explorer VI spectrum contains a larger number of electrons of higher energies than the spectrum of Explorer XII. It should be noted, that all of the electrons will be stopped in the space vehicle wall, but only those electrons below 0.3 MEV will be stopped in the space suit. This direct deposition of electrons in the space suit occupant will increase the absorbed dose since a human body will stop all electrons from 0.3 to 5 MEV. The absorbed dose rates due to electron deposition from the spectra of Explorer XII and VI behind a 1.5 gm/cm^2 aluminum shield would be 10 and 80 rad/hour respectively.

Another factor which should be considered in Bremsstrahlung production is the atomic number (Z) of the structural material. For the nylon-rubber extravehicular suit ($Z=8$, surface density = 0.08 gm/cm^2) the Bremsstrahlung dose rate is approximately 400 R/hour. The dose rates behind a comparable thickness of aluminum ($Z=13$) and steel wall ($Z=26$) for the peak electron flux in the Explorer VI spectrum are 650 R/hour and 1200 R/hour, respectively.

After adjusting for the surface densities of the walls considered, the dose rates for electron deposition and Bremsstrahlung were computed for the peak flux values and are shown in Table XXII. These calculated values agree with the dose

TABLE XXII
ELECTRON DOSE RATE

Atomic Number Z	Surface Density gm/cm ²	Outer Belt Peak Flux Dose Rates Rad/Hour		Lunar Trajectory Flux Dose Rates Rad/Hour	
		Electron Deposition	Bremsstrahlung	Electron Deposition	Bremsstrahlung
8 (nylon-butyl rubber)	0.08	1150.0	390.0	342.0	115.0
8 (nylon-butyl rubber)	1.5	130.0	18.0	43.2	6.0
13 Aluminum	1.5	100.0	30.0	33.0	10.0
26 Steel	1.5	20.0	60.0	23.5	20.0

rate values given by Russak⁽⁵⁹⁾. Also, the dose rate values for the lunar trajectory were found by numerically integrating the time electron flux relationship along the trajectory and then ratioing this value to the peak flux value. Calculations indicate that 1/4 inch of boron, plastic, or some other low Z material will decrease the Bremsstrahlung doses for a $Z = 13$ wall by a factor of 2 and for $Z = 26$ wall by a factor of 3. Therefore, a lower internal dose rates will result by covering the outer vehicle wall with a low Z material and increasing the amount of high Z material beneath the surface of the space suit.

d. Summary of Expected Internal Dose Rates

(1) Vehicle Occupant

An evaluation of the internal dose rate data in Table XXIII indicates that the vehicle or suit occupants and equipment will be subjected to the highest dose rates during the lunar trajectory. However, the total time for the lunar trajectory is of the order 160-200 hours, with only a small portion of time (4 hours) in the Van Allen belts where the dose rate is high. The total integrated dose of the vehicle occupant would be 40 rads. However, for the 24-hour orbit and the lunar site mission, the total accumulated dose for a year's operation in the vehicle would be a factor of 30 to 50 times higher than the dose for the lunar trajectory.

Also, when a vehicle is subjected to a one-year flight in a 550 km orbit, the dose from both the Van Allen belt and galactic cosmic radiation would be of the order of 10 rads. This low dose in the 550 km orbit is due to the orbit being below the near edge of the inner Van Allen belt.

(2) Space Suit Occupant

Any comparison of internal space suit-to vehicle doses should take into account the exposure duration, the surface density variation, and the different atomic number materials in these shields. Since the space suit is worn only for space maintenance tasks, the total dose can be minimized by judicious use of time during these tasks. The task time would be set by radiation dosage limit in comparison to the person's previous dose history. Therefore, for successful missions (i.e., no repairs in space) the dose would be set by the debilitation dose of the astronaut.

The surface density variation within limits, due to consideration of spectra of different type particles, can cause an absorbed dose change by a factor of 1.6 to 2 when comparing space suit wall to the vehicle wall. However, the surface density effect must be evaluated further by considering the shield materials, atomic numbers and their relative positions.

It can be concluded from calculations of the attenuation of space particles and secondary radiation that the most efficient shield is achieved by a low

TABLE XXIII
INTERNAL VEHICLE DOSE RATE (R/HOUR*)

	Suit**	Vehicle	24-HOUR ORBIT		LUNAR TRAJECTORY***		LUNAR SITE	
			Suit**	Vehicle	Suit**	Vehicle	Suit**	Vehicle
VAN ALLEN	3×10^{-3}	5×10^{-3}	1.9	1.2	16.0	8.0	0	0
GALACTIC COSMIC	2×10^{-3} to 2×10^{-4}	2×10^{-3} to 0.2×10^{-3}	3×10^{-3}	2×10^{-2}	1×10^{-2}	6×10^{-4}	1.3×10^{-2}	8×10^{-3}
	Depending on orbit in- clination.	Depending on orbit in- clination.						
MAXIMUM TOTAL DOSE RATE (VAN ALLEN AND GALACTIC COSMIC)	1.1×10^{-2}	7×10^{-3}	1.9	1.22	--	--	1.3×10^{-2}	8×10^{-3}

*R is defined as equal to roentgen, rad or rem

** Dose rate to which astronaut (not suit) is subjected

*** The man is exposed to the Van Allen belt radiation for approximately 4 hours during this trajectory.

atomic number (Z) material covering. As a result, the materials were selected to minimize radiation dosage due to secondary radiation by choosing a low Z material for external covering. The doses within a space suit expected for one day of continuous operation, are shown in Table XXIV.

TABLE XXIV

ESTIMATED DAILY DOSE RATES WITHIN A SPACE SUIT

Mission	Internal Suit Dosage (Rads)
Lunar Site	6.0
560 km Orbit	4.3
36,000 km Orbit	51
Repair in Lunar Trajectory (Inner Belt)	5.5 - 9.6
Repair in Lunar Trajectory (Outer Belt)	19.1 - 27.4

e. Predicted Radiation Effects

Two important classes of radiation effects must be considered during space maintenance activities: space system and component effects and the biological effects on the maintenance workers.

(1) Space System and Components

The total amount of radiation which any material receives is a function of the energy and flux of the radiation incident on the surface of the space vehicle. The nature and intensity of this radiation has been discussed earlier in this section.

Except for the surface exposed to the direct flux, the components and structural materials will not actually absorb much of the electron radiation since the penetration depth is very small. However, during the stopping process, Bremsstrahlung is produced. Depending on the thickness of the wall, little of this radiation (which is in the X-ray wavelength region), will reach the inside of the system. The protons, however, are more penetrating and some of them will deposit their energy in internal components and system parts. Also, by generation of secondary spallation products, the materials will be exposed to additional radiation from these products.

The important factor in determining the effect of radiation on component function is the ability of the material to perform properly after receiving a certain dose of radiation. Therefore, the important parameter is the total absorbed dose. The following list presents the electronic and electro-mechanical components which make up the space systems investigated during this contract.

ASD-TDR-62-931

Capacitors (tantalum, ceramic)	Accelerometers
Connectors (electrical cannon type)	Alternators
Crystals (rectifier, oscillator)	Scienoids
Diodes - Transistors	Generators
Condensers	Gyros
Reactances (choke coil)	Motors
Resistors (composition, carbon deposit, wire wound)	Relays
Insulators	Switches
Rectifiers (selenium)	Potentiometers
Transformers	Synchros
	Transducers
Tubes (gas, vacuum, Klystron, Magnetron, Thyatron)	Voltage Regulators

In addition to these components, the vehicles will employ various metals and metallic alloys as well as several different plastics and elastomers.

Table XXV lists doses at which the various materials and components would be appreciably altered. It should be noted that the threshold doses given in this table are conservative, consistent with the philosophy of approach already expressed. Thus, these doses may be increased for systems in which components and materials have been selected for their radiation resistance. Since the maximum anticipated dose rate behind a shield for the 550 km and 24-hour orbits or the lunar landing is approximately 10 millirads per hour, it would take approximately 100 years for the most sensitive material (viz. transistors) to degrade appreciably. Therefore, no radiation limitations are anticipated for nonviable material located within the vehicle. Metallics, metallic-alloys, or ceramics mounted on the outside of the vehicle or used for vehicle skin structure should be similarly unaffected. Two types of materials must receive special consideration; namely solid state devices mounted externally (e.g., infrared detectors or solar cells) and the space suit. Since the surface dose rate due to electrons may be as high as 10^6 times as that under only a few mils of material, it may be expected that for such devices, the surface layer will absorb a considerable quantity of ionizing radiation in relatively short times.

The space suit surface dose rates for the four missions are presented in Table XXVI. The times for 25 and 75 percent degradation of space suit materials are also shown in this table. Evaluation of these data to determine the length of time for space suit material failure indicates that the critical area would be a space maintenance task in the Van Allen belt during a trajectory. However, even in this worst case, the materials of the space suit (nylon and rubber) would be 75 percent degraded in

TABLE XXV

THRESHOLD DOSE* FOR VARIOUS MATERIALS

Material Or Component	Dose* (Rads)
Transistors	10^4
Glass	5×10^5
Teflon	10^6
Lucite	3×10^6
Butyl Rubber	2×10^7
Natural Rubber	3×10^7
Organic Liquids	10^8
Graphite	10^8
Polyethylene	10^8
Hydrocarbon Oils	5×10^8
Ceramic	10^{11}
Carbon Steels	2×10^{11}
Stainless Steel	5×10^{11}
Aluminum Alloys	10^{12}
Space Suit Materials	
Neoprene	10^6
Nylon	5×10^5

*Dose to produce appreciable alteration in a pertinent property or function.

TABLE XXVI

DEGRADATION OF EXTRAVEHICULAR SPACE SUIT BY RADIATION
DEPOSITION IN SUIT (RADS/HOUR)

	560 km Orbit	24-Hour Orbit	Lunar Trajectory	Lunar Site	
Due to Van Allen Belt	1.8×10^4	1.82×10^4	1×10^8	8×10^{-1}	
Due to Galactic Cosmic	2.0×10^{-4} to 0.2×10^{-4}	8×10^{-4}	1×10^{-4}	8×10^{-4}	
TIME TO DEGRADE SUIT (Hours)					
Due to Van Allen Belt	For 25% Degradation	3×10^3	3×10^2	0.5	6×10^6
	75% Degradation	3×10^4	3×10^3	50	
Galactic Cosmic	25% Degradation	3×10^7 0.3×10^7	6×10^6	5×10^7	6×10^6
	75% Degradation	3×10^9 0.3×10^9	6×10^8	5×10^9	6×10^8

50 hours. For the two considered orbits in the Van Allen belt, the time degradation is 3×10^4 hours for the near earth orbit and a decade lower for the 36,000 km orbit. Since for these particular time periods, the space suit occupant would have long since received a lethal radiation dose, suit material degradation is not a problem. For the same reasons, solar flare damage of space suit material should not be seriously considered, as again the space suit occupant would be dead long before the suit materials would be drastically degraded.

(2) Maintenance Worker

For any space system, the human occupant is the most radiation sensitive component. Thus, allowable biological doses will determine the radiation-limited extent of the maintenance task. Certain standards of permissible or tolerable dosages have been established by the National Committee on Radiation Protection (NCRP) for civilians working in radiation environments (51). For space missions, which presently are subject to many other nuclear hazards, the recommendations of the NCRP must be adjusted in view of the particular type of mission.

Biological dosages are measured in units of rems, where $\text{rem} = \text{RBE} \times \text{rads}$. RBE is the relative biological effectiveness of the particular type and energy of radiation for producing, in man, a given physiological effect. For electrons and X-rays (Bremsstrahlung), RBE is taken as unity. For heavy charged particles (e.g., protons), the RBE is a theoretical function of the LET, defined as the linear rate of energy loss of the particle in traversing a material. NCRP reports a maximum $\text{RBE} = 11$ for 0.1 MEV protons; and this value decreases to a $\text{RBE} = 1$ for protons of 10 MEV or higher energy. For purposes of this present study, an $\text{RBE} = 1.0$ has been assumed for all radiations to which the astronaut will be subjected.

The integrated doses to which the maintenance worker is subjected prior to recovery is difficult to specify because of the current controversy over acceptable doses. NASA has accepted the NCRP limits of 5 rem per year, 0.3 rem per week and 3 rem per quarter year as the limits for whole body radiation. A 25 rem limit has been accepted for emergency purposes. Pickering (52), on the other hand, has suggested that 25 rem is an acceptable mission dose for space flight with a 25 rem emergency increment. Schaeffer (49) estimates that a dose as high as 80 rem could be endured without serious consequences. Using Blair's formula and assuming 80 rem as the maximum allowable net dose, Schaeffer allows an accumulated dose of 765 rem in a three year period.

The performance decrement and clinical symptoms resulting from radiation exposure are presented in a number of research reports (53, 54, 55 and 56) and are reproduced in this report. Table XXVII presents the times required to receive 25, 100 and 300 rem for the four missions studied for an unshielded vehicle and a suited man outside the vehicle. We have not included the solar flare dose because of its random nature. Further, the man exposed to such radiation within either system

would be subjected to a fatal dose moments after the arrival of the high energy protons. It is important to note that the times required to reach a specified dose are essentially the same whether the man is protected by a vehicle or an extravehicular suit. Thus, from the point of view of radiation protection, the maintenance worker is not differentially protected by the vehicle or the space suit.

TABLE XXVII
TIMES REQUIRED TO RECEIVE SPECIFIC BIOLOGICAL
RADIATION DOSE DURING FOUR SPACE MISSIONS

Time — (Hours)					
	rem	560 km	24 hr	Lunar Traj.*	Lunar Site
Unshielded Vehicle	25	3,580	20.8	3.1-0.83	3,130
	100	14,250	83.2	12.5-3.32	9,350
	300	42,700	249.6	37.5-3.96	37,400
Space Suit Condition	25	2,270	13.2	1.56-0.34	1,925
	100	9,100	52.8	6.25-1.36	7,700
	300	27,300	158.4	18.7-3.96	23,100

Note: * Values are also the same of missions of 24 hour orbit and lunar site.

Table XXVIII presents an estimate of the total number of days per month that a space maintenance worker could be expected to spend in earth orbit (e.g., 560 km orbit) before reaching an arbitrary integrated dose of 90 rem due to solar flares alone. This estimate was derived from Reference 49 for shield thicknesses of 1.5 gm/cm² and 0.08 gm/cm². The latter value represents the case of an astronaut outside the vehicle in a nylon-neoprene space suit. In addition, an unshielded dose per flare of 433 rem has been assumed.

TABLE XXVIII

560 km ORBIT DURING SOLAR FLARES

No. Flares/Mo.	Mission Duration (Days)	
	0.08 gm/cm ² (Space Suit)	1.5 gm/cm ² (Unshielded Vehicle)
1	6.3	11.0
2	3.1	5.5
3	2.1	3.7
4	1.5	2.7

It should be noted that all four missions under investigation will at some point encounter this type of radiation. In general, the solar flare radiation may be assumed to be isotropic and omnidirectional, since it emanates from the sun. However, at certain orbital inclinations and juxtapositions of earth, moon, and sun, the vehicle or suited man will be shielded completely or partially from direct solar flare radiation.

D. VACUUM

1. Description

The concept of the hydrostatically supported atmosphere within a few hundred kilometers above the earth's surface was based, in the past, on assumed thermodynamic temperatures of only a few hundred degrees centigrade. As a result, a sharp decrease of density within this altitude range was assumed. Similar considerations were applied to the exosphere, defined as the layer above the critical level where the mean free path of a gas particle equals approximately 0.5 of the local decimal scale height.

This concept is, at present, completely revised and, in addition, altitude ranges up to several thousand kilometers have been explored in terms of gas density and composition, electron density and electron ion temperature. It is recognized that high temperatures prevail in the exosphere (perhaps 1200°K) near the top of the F-layer. Atomic oxygen diffusing upwards from the ionospheric heights, dominates to a height of perhaps 1200 km (57). Atomic hydrogen forms a minor constituent below this height. Its concentration, however, decreases much more slowly with increasing altitude, and as a consequence, it will dominate the particle distribution at altitudes exceeding 1200 km.

Different observing techniques were used to determine the electron distribution. They provided some useful information, however, there still exist uncertainties in the interpretation that would result in variations of the distributions not representative in the exosphere. In particular, a better knowledge of the electron distribution between 1000 and 1500 km at various latitudes, would provide a better background for the description of the proton distribution of the outer ionosphere.

From the assumed temperature, the molecular weight and density, a prediction of the pressure-altitude function of the exosphere and ionosphere can be attempted. Table XXIX shows values of pressure, temperature, concentration of molecules and composition at altitudes between 200 and 20,000 km. These values were based on the evaluation of 18 publications on the exosphere and ionosphere and is published in Reference 25.

It can be seen from this table that the low pressure values encountered by space systems range from 10^{-6} mm of Hg (corresponding to a 200 km orbit) to values less than 10^{-12} mm of Hg (corresponding to a 24-hour orbit).

2. Behavior of Material in High Vacuum

The low pressure to which space systems or space workers inside a capsule or suit will be subjected during the different operational phases, might result in several effects on the material used for the structure, propulsion and control systems, manipulators and tools. These effects are reviewed in the following section.

TABLE XXIX

IMPORTANT QUANTITIES OF THE EXOSPHERE

Altitude	Pressure mm Hg	Temperature °C	Concentration molecules, atoms or ions/cm ³	Composition
200 km	10 ⁻⁶	10 ³	10 ¹⁰	N ₂ , O, O ₂ , O ⁺
800 km	10 ⁻⁹	10 ³	10 ⁶	O, O ⁺ , H
6500 km	10 ⁻¹³	10 ³	10 ³	H ⁺ , H
Above 20,000 km	10 ⁻¹²	10 ³ to 10 ⁵	10 ¹ to 10 ²	85% H ⁺ , 15% H ⁺⁺

a. Loss of Inorganic and Organic Material

The evaporation of material can be predicted quite reasonably from the Langmuir equation. Its application to metals such as Al, Be, Cr, Fe, Si, Ni, Co, and Ti, and to the more chemically stable oxides yields very low surface recession rates not exceeding 10^{-5} inches per year at temperatures between zero and 1000°F .

Organic materials such as plastics and rubber are much more susceptible to a high vacuum even at moderate temperatures, either due to vaporization or decomposition followed by vaporization. Many of these materials are high molecular weight polymers and as such, break down relatively easily into their component monomers. This process might be accelerated by particle radiation. Since the molecular weights or fragments of the deteriorated molecular chain are unknown, direct measurements of the surface loss are preferable.

In addition, decomposition rates of certain polymers are accelerated by small amounts of impurities or additive agents. Catalysts added to induce polymerization of many monomers may promote decomposition of the polymer. Plasticizing agents and mold lubricants used to modify mechanical properties and as aids in fabrication are also generally detrimental to material stability.

Vacuum stability might be improved by certain degradation inhibitors or surface coats. Polymers are also more stable at elevated temperatures in the absence of oxygen.

b. Property Changes of Inorganic and Organic Materials

From a review of results obtained on the change of mechanical properties of inorganic materials, no significant decrease under high vacuum conditions can be anticipated unless the temperature approaches a value at which appreciable evaporation takes place. Some investigators found a measurable effect of the fatigue life for nickel-based alloys and also for stainless steel. The effect may be positive or negative (increase or decrease of creep strength), depending on whether the tests were performed at low or high temperatures.

Weight loss of organic material may be accompanied by significant changes in mechanical and electrical properties, as well as in dimensions. It may be said, in general, that weight losses of one or two percent do not produce property changes of design importance. Weight losses of ten percent per year, however, will result in a considerable change in mechanical and physical properties, in particular, at elevated temperatures.

Plastics and elastomers have to be excepted from this rule. These materials, in particular, those containing external plasticizers, may embrittle significantly after weight losses of less than ten percent and should be generally avoided in a space system design.

c. Sealing, Friction, and Lubrication

Space systems and also space capsules might incorporate external parts. Also electrical cables for power supply to external components, electromotors, or sensors might be required. In these cases, the leakage through the seal shall not exceed a permissible amount, even under long time exposure to the vacuum in space and under temperatures ranging from -250°F to 300°F . In connection with the sealing problem, friction and lubrication must be considered. The self-cleaning process of surfaces subjected to high vacuum reduces the lubrication property of surfaces moving over one another. Whether this will result in the so-called cold welding, depends on the ratio of the interface area being in direct contact to the total surface area, and also on the properties of the material sliding against each other.

A vacuum-tight seal around the moving part will provide maximum protection, particularly for low speed operation, but at the expense of frictional loss. Low vapor pressure oil and grease (for example, silicon base) should be used. Also, the selection of solid lubricants (characterized by an evaporation loss less than that of liquid lubricants) might be considered; for example, Molybdenum-disulfide bonded with silicone, phenolic or an epoxy resin.

Very thin layers of gold and silver, operating against a harder metal, have been successfully used in X-ray tubes at 10^{-6} mm of Hg over long duration. Similar results were reported from Tetrafluoroethylene sliding against steel.

3. Summary

In summarizing the present results obtained on material in high vacuum tests, it can be said that no particular problem can be foreseen in the use of metals, alloys, or oxides of the more chemically stable configuration, even under long exposure. Consequently, no repair or maintenance tasks caused by high vacuum on the overall structure of space systems or capsules are anticipated, if proper selection of material is assured. For example, aluminum and its alloys are very suitable because they combine high vacuum stability with good particle radiation and micrometeoroid shielding properties.

Most organic materials pose problems for several reasons. We know that many of them are much more susceptible, but considerable less data are available about their behavior. In addition, all laboratory experiments in organics were conducted in a simulated high vacuum that did not exceed 10^{-8} mm of Hg.

As shown in the introduction of this section, the pressure at altitudes above 800 km exceeds 10^{-9} mm of Hg and decreases by several decades above 20,000 km. Even an experimental evaluation of a particular organic material in a vacuum that can be maintained over long exposure time in a test chamber, does not permit a definitive conclusion as to the same behavior at the much lower pressure in space and during

exposures over years. Consequently, the application of certain organic materials might result in unexpected maintenance and repair tasks, particularly, for moving surfaces requiring good sealing and lubricating.

CHAPTER 6

MAN'S CAPABILITY TO PERFORM SPACE MAINTENANCE TASKS

A. GENERAL

The broad spectrum of conditions in the space maintenance complex that has been presented in the foregoing analysis of the ESWS, the MTSS, SLOMAR, and the Lunar Vehicle defines, in a preliminary fashion, the situation in which space maintenance workers must operate and the tasks they must perform. The major objective of this section is an assessment of the potential of man to utilize his unique ability to implement corrective maintenance procedures as needed, under the environmental conditions expected in space. The approach to be followed in attempting to meet this objective is basically one of describing the tasks, to the extent possible, in behavioral terms and then evaluating the ability of the man to accomplish these behaviors within the environmental and temporal limitations.

The effects of radiation and micrometeorites on the biological well being of the maintenance worker are discussed in Chapter 5, whereas the interactions between tools, fasteners and remote manipulators and the capabilities of the maintenance worker are discussed in Chapters 8 and 9, respectively.

B. TASK REQUIREMENTS

From a review of the maintenance requirements that were defined in the maintenance analysis chapter, it was possible to make a rough determination of the extent to which the various subsystems would require maintenance external to the vehicle and the type of activities required. These data are presented in Table XXX.

It is of some interest to note that only the propulsion and structure areas are expected to involve predominantly external activities. These two areas also represent ones on which actual repairs may have to be effected in lieu of replacement of components and assemblies. These requirements are summarized by the space systems in Table XXXL

Although the maintenance tasks have been defined in an earlier section, in terms of sequence of equipment, related activities, to determine the full import on the man an attempt was made to define the tasks in terms of the specific behaviors required to accomplish them.

Thus, representative tasks have been selected from the task analysis completed during the study as being typical of the maintenance requirements that may occur in space. An interior maintenance function, for example, might be in replacing a fuel cell in the lunar landing vehicle. Exterior functions might include the following:

TABLE XXX

DISTRIBUTION OF MAINTENANCE ACTIVITIES INTERNAL AND
EXTERNAL TO THE VEHICLE

Subsystem Area	Predominant Maintenance Requirement	Task Locations (%)	
		Intravehicular	Extravehicular
Rendezvous and Docking	Component Replacement	50	50
Assembly	Component Replacement	50	50
Propulsion	Leakage Correction Contamination Component Replacement	20	80
Avionics	Component Replacement	90	10
Structures	Inspection Damage Repair	25	75
Power Supplies	Leakage Correction Component Replacement	90	10
Life Support	Leakage Correction Contamination Component Replacement	95	5

TABLE XXXI

PRINCIPAL MAINTENANCE TASKS

Space Vehicle	Task	
	Internal	External
1. Logistic Vehicles	<ol style="list-style-type: none"> 1. Leak correction in plumbing 2. Component troubleshooting, inspecting, adjusting, calibration, repair and replacement 	<ol style="list-style-type: none"> 1. Leak correction in plumbing 2. Repair of meteor damage 3. Troubleshooting, inspection and replacement of components
2. ESWS and MTSS	<ol style="list-style-type: none"> 1. Assembly and activation 2. Leak correction in plumbing 3. Component troubleshooting, inspecting, adjusting, calibration, repair and replacement 4. Bench testing, trouble-shooting, and repair 5. Repairs to structure 	<ol style="list-style-type: none"> 1. Assembly 2. Leak correction in plumbing 3. Troubleshooting, inspection 4. Repairs to structure and external components 5. Logistic services

- (1) Replacing an IR sensing receiver assembly on the logistics vehicle.
- (2) Replacing a leaking gas generator gasket on the logistics vehicle while docked on the ESWS.
- (3) Replacing tracking transponder/components of the ESWS.
- (4) Repairing meteor punctures in the outer wall of the ESWS.

From review of the tasks presented in Tables VII through XII the following list of extravehicular behaviors has been derived:

- (1) Locomotion within the vehicle, unsuited.
- (2) Donning space suit.
- (3) Locomotion within the vehicle, suited.
- (4) Suit communication equipment checkout.
- (5) Mounting maneuvering system.
- (6) Maneuvering system checkout.
- (7) Selection and securing of tools and components.
- (8) Securing and checking tether cable.
- (9) Suit pressurization and airlock evacuation.
- (10) Airlock egress.
- (11) Locomotion outside the vehicle.
- (12) Workplace positioning and securing.
- (13) Torqueing with screwdriver or wrenches.
- (14) Component positioning and securing.
- (15) Torqueing while holding a hand grip.
- (16) Joining - torqueing by hand, with a screwdriver, and/or wrenches.
- (17) Visual checking.
- (18) Joining-drilling, riveting, welding, sealing.
- (19) Airlock ingress
- (20) Airlock pressurization and check - suit depressurization.
- (21) Doffing space suit and return to station.

It becomes apparent that all conceivable varieties of task elements are called for. As a minimum, the following behavioral task elements will be involved:

- (a) Search - phase in which eyes or hands are groping for an object.
- (b) Find - terminates the search as a mental reaction rather than a physical.
- (c) Select - pick out from among several objects for work.
- (d) Grasp - picking and holding preparatory to manipulation.
- (e) Transport Loaded - moving of an object (hands-fingers, carrying, sliding, dragging or pushing).
- (f) Assemble - hand moves the piece into its place, assembles and releases the part.
- (g) Use - manipulating a tool device or piece of apparatus.
- (h) Disassemble - begins when the hand starts to move one part from the assembly and ends when the hand has separated the part completely.
- (i) Inspect - a mental behavior which may occur together with other activities in testing for compliance with standard size, shape, color or other sensory quality.
- (j) Pre-position - positioning an object in such a predetermined way that it may be grasped in the position in which it will be held when needed.
- (k) Release load - phase in which the hand is letting go of the load.
- (l) Transport empty - moving the empty hand in reaching for an object.

Although the internal tasks will represent a substantial portion of the maintenance activity, the external requirements were given considerable attention because they pose some special problems in assuring their fulfillment. Thus, a detailed analysis of specific behaviors in two task areas was accomplished; one on a structural task and one on a propulsion task. Data from this analysis are given in Tables XXXII and XXXIII. It should be noted that these task listings and the times estimated to complete them are based on the assumption that the malfunctioning components are readily accessible to the maintenance worker. From the data in Table XXXIII it will be noted that, 18 task units requiring discrete reaching movements have been identified. The total number of reaches is estimated to be approximately 100. To accomplish this task within the time estimated in the task analysis will require an arm extension or retraction, on the average, every 14 seconds. Thus, if the pressure suit offers any appreciable resistance to bending at the joints, the worker will be required to exert a considerable amount of energy in body movements alone.

An even more important consideration, however, is the requirement for grasping and holding. It will be noted that 34 task units requiring grasping and retaining have been identified. Again, as in the case of reaching, the actual number of movements is substantially greater than the task units; in this case being in excess of 100. This number of actions becomes even more important when we note that approximately one-half of them involve small nuts and bolts (3/16 inch). In a study by Peters and Mitchell⁽⁶²⁾

it was found that considerably difficulty was encountered handling 9/16 inch nuts, which have approximately twice the outside diameter of the 3/16 inch nuts, in a pressurized suit. Thus, it appears that, unless substantial improvements in the manipulating capability of space suits are made, the performance of tasks such as this will be extremely difficult for the worker to perform. The alternative would be redesign of the equipment to provide larger or different kinds of fasteners. This, however, places a severe burden on systems design and of course would be impossible for the current generation of space vehicles.

Another consideration on a task such as this will be the length of time required to complete the job. It will be recalled from the task analysis that the elapsed time for the gas generator repair task was approximately 24 minutes. There is, however, no degradation built into these time estimates for pressure suit restrictions. If we apply a degradation of 100 percent as is discussed later in this section and by the data from Peters and Mitchell (62), the time for performance of this task now becomes about 48 minutes plus the time required for translation to and from the work area, positioning and tethering time, etc. Because of the large number of small work pieces that must be handled, however, the task performance time will undoubtedly be substantially greater than that mentioned above. An alternative approach to the completion of the task may be to disconnect the malfunctioning unit and return to the interior of the vehicle for its repair. Additional analyses will be required to determine the extent the additional expenditure of energy and its associated increase in life support requirements would be offset by the losses associated with the additional ingress and egress and by the increased propellant requirements for the workers translation system.

X - Required
P - Possible Requirement

TABLE XXXII
BEHAVIORS REQUIRED FOR STRUCTURAL REPAIR
(Micrometeorite Puncture)

Task Element	Translation			Arm/Hand Movement			Force Adjustments					
	Walk	Crawl	Space	Reach	Grasp	Hold	Gross	Fine	Pull	Push	Switch	Tether
Don Suit	X			X	X	X	X	X	X			
Checkout Suit								X			X	
Secure Tools					X		X		X		X	
Enter Airlock	X	X										
Secure Hatch				X	X		X		X			
Secure Tether				X	X			X				
Pressurize Suit and Checkout					X			X			X	
Depressurize Airlock							P	P	P	P	P	
Open Outer Hatch	X	X		X	X		X					
Checkout Propulsion Unit				X	X		X	X			X	
Egress			X						X	X	X	
Maneuver to Work Area			X					X	X	X		
Position on Wall	X	X		X	X	X	X	X	X	X	X	
Secure Tethers	P	P		X	X	X	X	X	X	X	X	
Prepare cutting tool					X	X		X	X	X	X	
Cutout Damaged Area					X	X	X		X	X	X	
Collect Debris				X	X		P	P	P	P	P	
Remove Damaged Panel				X	X	X	X		X	X		
Secure Tools				X	X	X			X	X		

Required
Throughout
Outside
Operation

TABLE XXXII (CONT)

Task Element	Translation			Arm/Hand Movement			Force Adjustments					Tether	
	Walk	Crawl	Space	Reach	Grasp	Hold	Gross	Fine	Pull	Push	Switch		
Apply Template				X	X	X		X		X			X
Prepare Drill Tool				X	X	X	X		X	X			
Drill Holes					X	X	X	X		X		X	
Secure Tool				X	X	X			X	X			
Remove Template/Store				X	X			P	X				
Seal/Repair Coolant Coils				X	X	X	P	P	P	P		P	
Apply Sealing Compound				X	X	X			P	P		P	
Insert Prefab. Plug					X		X			X			
Insert Fasteners (rivets)						X	X			X			
Upset Rivets-Squeeze					X		X		X	X		X	
Release and Store Shear Pin					X	X		X		P		X	
Secure Rivet Tool						X						X	
Detach Local Tether/Tiedown	P	P		X	X		X			X		X	
Return to Airlock			X					X		X		X	
Ingress	P	P	X						X	X		X	
Deactivate Propulsion Unit							X					X	
Close Outer Hatch	X												
Pressurize Airlock													

TABLE XXXII (CONT)

Task Element	Translation			Arm/Hand Movement			Force Adjustments					Tether
	Walk	Crawl	Space	Reach	Grasp	Hold	Gross	Fine	Pull	Push	Switch	
Open Inner Hatch Enter Station	X						X					
Totals (While at work area tether in place)			12	17	13		11	7	12	15	10	

TABLE XXXIII

BEHAVIORS REQUIRED FOR REPLACEMENT OF GAS GENERATOR INJECTOR GASKET

Task Element	Translation			Arm/Hand Movement			Force Adjustments				Tether	
	Walk	Crawl	Space	Reach	Grasp	Hold	Gross	Fine	Pull	Push		Switch
(Worker-outside-with all equipment and parts-tethered at work place)												Req'd. Through-out
(Secure Ignitors)												
1. Cut lock wire on connector (4)				X	X	X						
Stow Tool					X		X					
Separate Electrical Connectors (4)					X	X						
Stow Tools				X	X	X						
(Remove 3/16" Bolts From Clamps (2)												
Torque nuts and remove					X	X	X		X	X	X	
Remove Bolts					X		X	X				
Stow Nuts and Bolts or				X	X	X						
Replace Nuts and Bolts in Unattached Clamp				X	X	X		X	X	X		
(Remove 3/16" bolts from Bracket (2)												
Torque Nuts and Remove					X	X	X		X	X	X	
Stow Nuts				X	X	X						
Remove Bolts and Tapered Spacers					X		X	X				

TABLE XXXIII (CONT)

Task Element	Translation			Arm/Hand Movements				Force Adjustments				Tether
	Walk	Crawl	Space	Reach	Grasp	Hold	Gross	Fine	Pull	Push	Switch	
Stow Bolts and Spacers				X	X	X					X	Req'd. Throughout
Stow Tools				X	X	X						
(Disconnect Flex Hoses From Gas Generator Injector												
Torque Nuts					X	X	X		X	X		
Stow Tools				X	X	X					X	
(Remove Bolts From Flange Joints)												
Torque Nuts and Remove					X	X	X		X	X		
Stow Nuts				X	X	X						
Remove Bolts					X	X	X	X				
Stow				X	X	X						
Stow Tools				X	X	X					X	
(Remove Gas Generator Starter Assembly)												
Pry Apart					X	X	X		X	X		
Stow Assembly				X	X	X					X	
(Remove Gasket)												
Pry out with Thin Blade					X	X	X	X	X	X		

TABLE XXXIII (CONT)

Task Element	Translation			Arm/Hand Movements			Force Adjustments					Tether
	Walk	Crawl	Space	Reach	Grasp	Hold	Gross	Fine	Pull	Push	Switch	
Collect and Stow Debris				X	X	X						Req'd. Through-out
Install Gasket				X	X	X		X				
Replace Gas Generator Assembly				X	X		X	X	X	X		
(Replace 10 Bolts in Flange Joint)												
Insert Bolts					X	X		X	X			
Place Nuts					X	X		X				
Torque Nuts					X	X	X	X			X	
Connect Flex Hoses					X	X	X	X	X	X		
Torque Connection							X	X	X	X	X	
Replace 3/16" Bolts in Bracket				X	X	X	X	X	X	X	X	
Replace 3/16" Bolts in Clamp				X	X	X	X	X	X	X	X	
Replace Electrical Connections				X	X	X	X	X	X	X		
Secure Lockwire in Connectors					X	X		X	X	X		

C. MOTION AND MANUAL DEXTERITY

Parameters affecting the man's capability to perform the maintenance tasks include his range of motion, dexterity, and performance facility at the work area. These in turn are affected by where he is performing and the protective equipment he may be wearing. Disregarding for the moment the effects of such factors as weightlessness, subgravity or coriolis, the concern here is with his immediate work-space requirements and manual-dexterity capability, and how these might be affected by protective equipment.

Voluminous data have been compiled on the human motor system in a normal environment. The human motor system may be described as an assortment of tendons, ligaments, muscles, sensory organs and nerve networks. The system is made up of articulated body members that can be moved through a range of angular arcs. Data, such as that presented by Ely (63) on see and reach capabilities, with respect to a seat-reference point either standing or sitting must be considered in the layout of work areas.

Data on body dimensions in kneeling, crawling or prone working positions for the 95th centile of a population sample developed by Hertzberg and others (64) should be considered in the design of maintenance access areas within the vehicle. This may become of particular interest with regard to intravehicular tasks because of the limited volume of space vehicles and in the design of exit areas of the vehicles.

Since speed and accuracy of manipulation actions will also affect the effectiveness with which a man can perform, these variables must be considered in defining the capability of the astronaut. Data are presented in Figure 43 (after Brown and Slater-Hammel, (65)) and Figure 44 (after Brown, et al, (66)) which define speed and accuracy for hand movements in positioning a sliding device.

Rubin and others (68) employing a Universal Motion Analyzer, determined that vertical motion sequences required shorter travel time than horizontal but manipulation was the same. It was also noted that type of manipulation has an influence on travel time; for example, the travel time required for switch turning was 32 percent faster than for pin pulling. Other studies have also confirmed this. (68)

Manipulatory skill has also been studied relative to design criteria. Davis (69) found that times required in making settings under zero-torque conditions were shortest for small cranks and wheels (e.g., one to six-inch radii); under torque conditions of 40 or 90 inch-pounds the larger radii were superior. Reed (70) studied the speed-of-rotation capabilities of the human operator using various radii of crank handles under zero torque conditions. Results are presented in Figure 45. Stump (71) found that finger dexterity in using 1/4 inch diameter knob, to extinguish a neon light by balancing a bridge circuit, was as efficient, and in some cases more so, than in hand or wrist action using a 3/4- and 2-inch diameter knob. Subjects tended to roll the 1/4 inch knob between thumb and forefinger, and this was true whether bare handed or wearing gloves. (72)

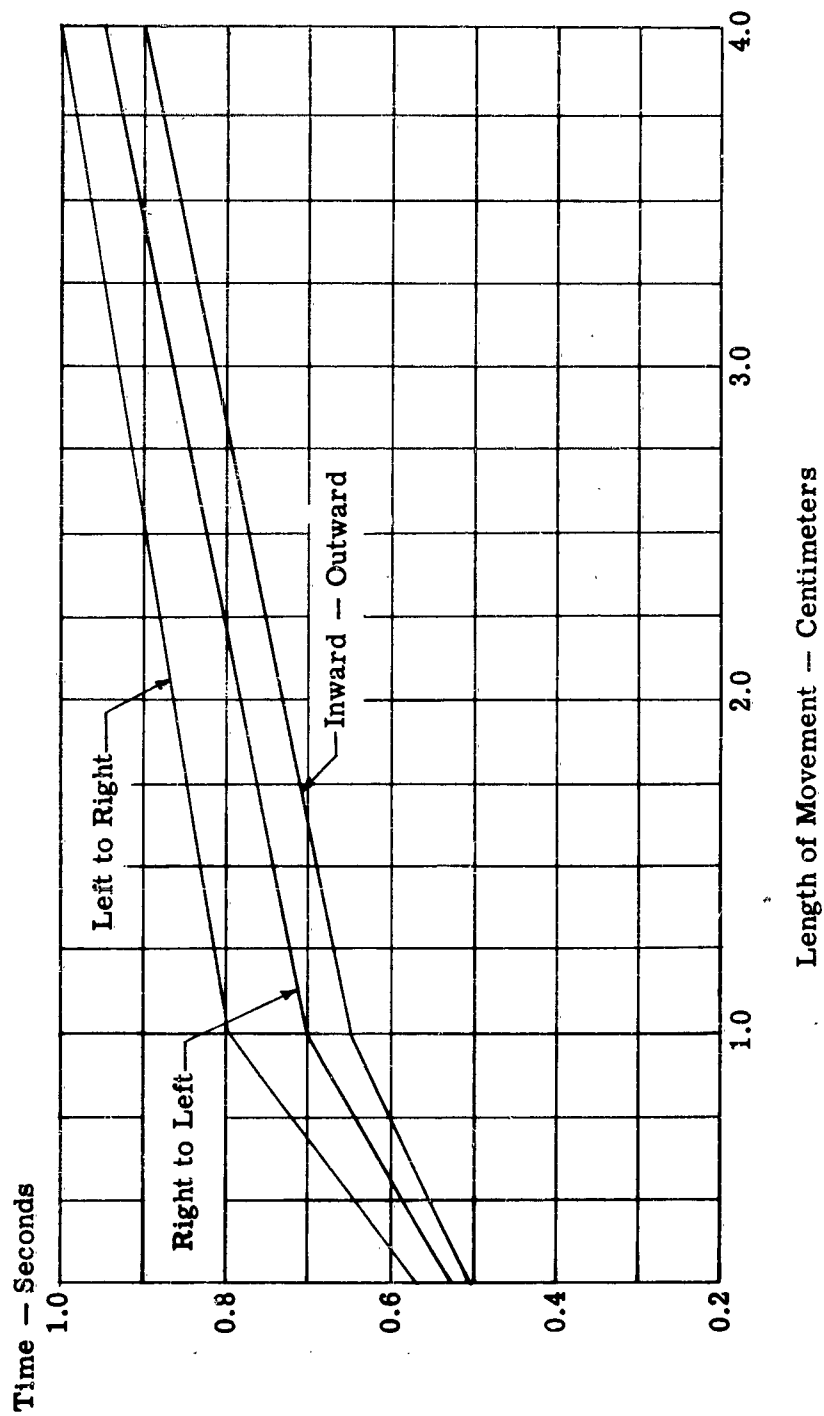


Figure 43. Time Required to Execute Positioning Movement
(After Brown and Slater and Hammel (65))

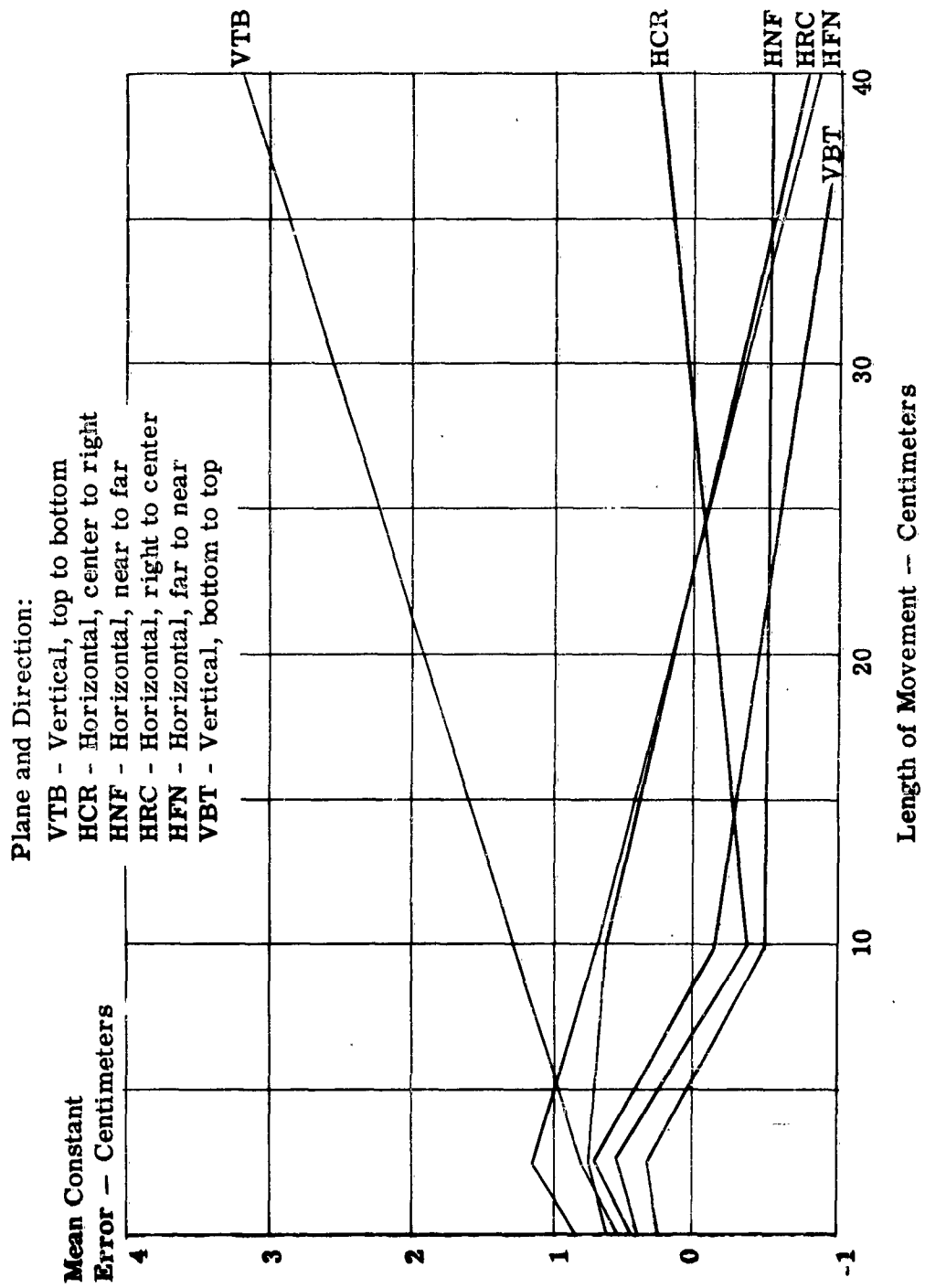


Figure 44. Accuracy in Executing Position Movements (After Brown, et al, Reference 66)

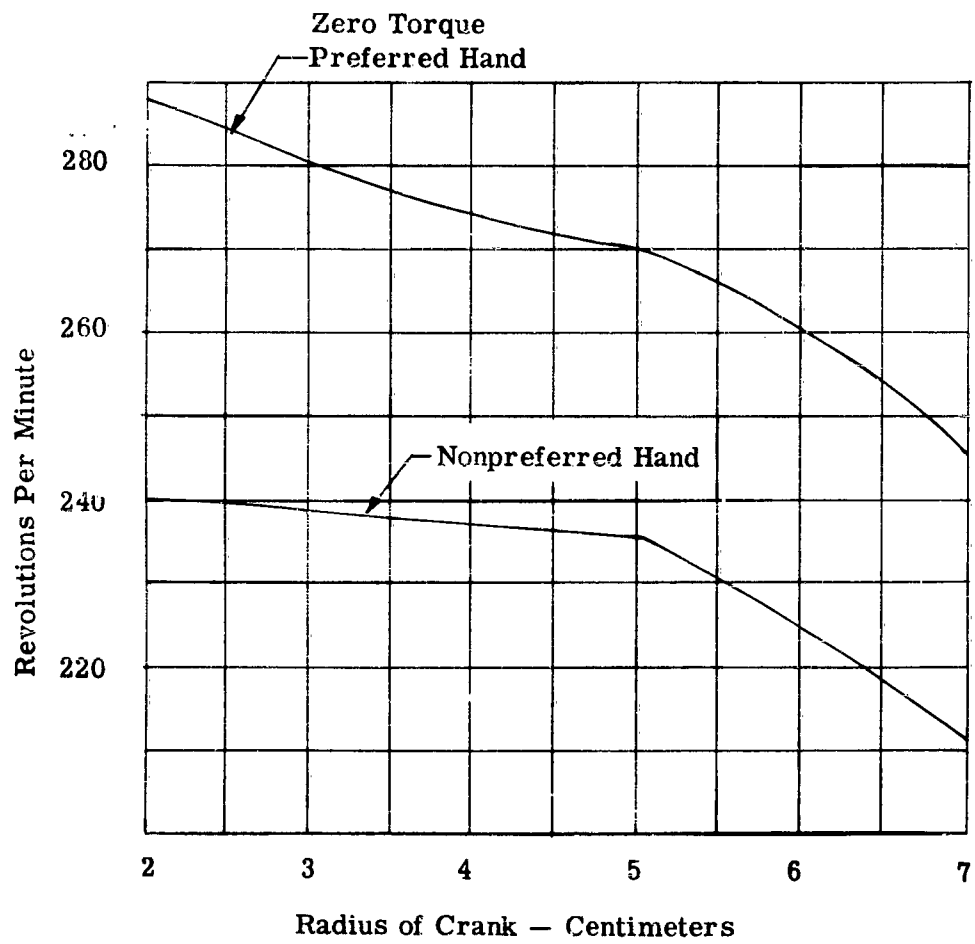


Figure 45. Speed of Cranking with Respect to Radii. (After Reed, 70)

TABLE XXXIV
ARM STRENGTH DATA IN NORMAL ENVIRONMENT
FOR 5th CENTILE (IN POUNDS)

ARM ANGLE (DEGREES)	PULL		PUSH		UP		DOWN		IN		OUT	
	L	R	L	R	L	R	L	R	L	R	L	R
180	50	52	42	50	9	14	13	17	13	20	8	14
150	42	56	30	42	15	18	18	20	15	20	8	15
120	34	42	26	36	17	24	21	26	20	22	10	15
90	32	37	22	36	17	20	21	26	26	18	10	16
60	26	24	22	34	15	20	18	20	17	20	12	17

A study by Hunsicker⁽⁷²⁾ has provided data on arm force-application capability in adduction, abduction, flexion and extension. Data that are pertinent to equipment design for maintenance activities are presented in Table XXXIV.

Hugh-Jones (73) has studied leg force capability. When the leg is extended to an angle of beyond 160 degrees, exertable force markedly declines. Average maximal exertable leg forces for three groups of subjects were determined to be 845, 691, and 689 pounds, respectively.

Fisher and Berren (74) have presented data on hand-strength tests of a male industrial population, where the average occurred at about 120 pounds of grip, and ranged from around 70 to 175 pounds. Other studies, e.g., Clarke (75) have provided data on back-lift capabilities of a physically conditioned population. In back lift, i.e., lifting with the legs straight, he found the range to vary from 330 to about 800 pounds, the average being 520 pounds. Leg lift capability, i.e., starting with legs bent, varied from 700 to 2500 pounds, with the average being 1480 pounds. It would appear from the foregoing that the man is sufficiently strong to accomplish almost any maintenance task in space if his energies can be harnessed. Unfortunately his capability to apply these forces is considerably less in space than on earth.

In an investigation conducted by Celantano and Alexander (76), force capabilities were studied while sitting on a near frictionless air bearing stool, as compared with sitting on the stool in normal friction. Force capabilities were as follows:

<u>Force Application</u>	<u>Normal (Pounds)</u>	<u>Air Bearing Frictionless (Pounds)</u>
Push	73.4	4.8 (95.0 momentary)
Pull	61.1	2.3 (118.4 momentary)
Compression	102.6	100.3
Extension	93.7	92.3
Torque, push	47.5 ft. lbs.	16.7 ft. lbs. (43.3 momentary)
Torque, pull	47.1 ft. lbs.	25.7 ft. lbs. (37.5 momentary)

It was found that even with a handhold, sustained force was impossible. It will be recalled from Chapter 4 that torques on the order of 25 to 30 pounds would be required for some of the tasks analyzed. In the investigation of the effectiveness of various standard tools it was found that with the use of a lathering arrangement and a handhold substantial torques could, indeed be applied in an acceptable amount of time. Table XXXV illustrates some mean times required to accomplish both torqueing tasks on an air bearing floor.

TABLE XXXV

**RESULTS OF TESTS CONDUCTED ON AN STANDARD
AIRCRAFT BOLTS UTILIZING THE AIR BEARING PLATFORM**

Bolt Size	Mean time to fasten in seconds			Mean time to unfasten in seconds		
	Open end Wrench	Box Wrench	Socket Ratchet	Open end Wrench	Box Wrench	Socket Ratchet
AN-3	23.6	19.1	19.5	19.4	21.0	19.4
AN-4	42.8	33.4	25.0	34.1	31.2	23.8
AN-5	30.4	28.0	29.2	27.2	31.0	24.1
AN-7	25.0	27.1	24.8	31.5	29.6	23.6
AN-8	35.1	34.3	31.7	32.2	33.1	29.0
AN-12	47.2	-	31.6	42.5	-	39.7
AN-14	63.	-	-	52.5	-	-

NOTE: The times indicated for AN4, 5, 7, and 8 were taken with subject employing a handhold. Those for the AN12 and 14 employed both a handhold and a foot-braced tether. The blanks result from lack of an appropriate tools in that size category.

Study of the above data, in relation to maintenance, suggests that the primary problem with regard to intravehicular tasks is maintaining position, in so far as motor activity is concerned. There seems little reason to assume any substantial losses in dexterity and mobility, as long as the man works in a shirt sleeve environment. Thus, if the maintenance worker is able to anchor himself without undue restriction, he should be able to perform a majority of the internal tasks that do not require use of a pressure suit or other means of encapsulation with approximately the same ease (or difficulty) that he can on earth. Mueller (77) has shown, for example, in Zero-G-flights in the USAF C-131 B aircraft that if a man is securely tethered to his work area and maintained at a desired distance from the work object the a brace, that his performance is not significantly reduced over what it would be in a one-g environment.

Analysis of the task and behavioral data as it relates to performing maintenance functions external to the vehicle, suggests that man's capability to do many of these tasks will be reduced because of certain environmental factors. Perhaps the more important ones are the requirement for protecting the worker from the low vacuum of space and the uniqueness of the perceptual and visual stimuli he will encounter. The following sections discuss these problem areas.

D. EXTRAVEHICULAR SUIT RESTRICTIONS

The design and development on a type of suit to be worn by the space maintenance worker, has only recently been undertaken. A brief description of some of these programs are given in the following paragraphs. It will be noted that one of the major considerations in each case is the provision of adequate mobility and dexterity.

David (78) describes a space suit offering almost complete mobility undergoing design for the Air Force by the International Latex Company. The suit is planned for adaptable prolonged usage outside the space vehicle, on the moon, and for interplanetary travel. Joints of the suit are to be intricately convoluted with no heavy metal gimbals nor re-inforcing straps. It will have a total weight of 21 pounds, easily and quickly donned with no help. Pressurization is to be provided by pure oxygen to 5 psi from a source within the vehicle or a back pack carried on the suit. Under full pressure, the user is expected to have tactile sensitivity and finger dexterity. The structure is of aluminized fabric, nylon and neoprene, welded at the joints. To promote dexterity in the hand, a substantial area in the palm is kept flat during inflation by a fiberglass insert imbedded in the surface of the glove. The fingers, with their small cross section, offer little resistance to bending even without any deliberate assistance in the form of folds or bellows.

Alexander (79) has described a NASA space-suit development, sufficiently light and mobile to be worn as personal equipment, and adjustable to any individual's form. An aluminum welded grid frame attaches to a mobile couch and restraint system for acceleration protection. The suit, it is expected, may eventually find application on multimanned, long-duration space flights.

Belasco (8) has described the General Electric development of a space CAPE (Complete Anthropometric Protective Enclosure.) This is to be a space suit designed for four hours maximum wear outside the space craft, or eight hours within. The suit would provide protection for temperature extremes in solar radiation, reflection and emission from the vehicle and the extremely low temperatures in the shadows of space; radiation protection is to be provided for, including a warning system and abort procedures when indicated to be above critical levels. The helmet would include a protective visor, provisions for feeding and vomitus removal and buffeting protection. The Space CAPE materials and configurational design criteria were described in addition to an activities analysis of the predicted task behaviors required of the space maintenance worker. These include such things as finger dexterity in plugging connections, attaching/detaching tools from his belt and attaching lines. Other manual activities are designated, such as squeezing, positioning, and pointing a pressure sealant gun, locomoting by hand-over-hand action, and torquing wrenches. He concludes from the feasibility study that pressure suit mobility commensurate with the task requirements appears attainable.

A major design and development study of an extravehicular suit is currently underway at the Hamilton Standard Division of The United Aircraft Corp., under sponsorship of NASA. Design information on this system was not available at the time of the conduct of this study, however.

Although little solid experimental data is available on mobility and performance capabilities in a pressurized suit, examination of what is available, in conjunction with the expected configurations of advanced space suits, should provide some insights into the effects these suits will have on the performance of the wearer.

In the Mark II full measure suit study (81) an F8U-1 aircraft was employed, using motion pictures, interviews, questionnaires and direct-observational data for evaluation. Control access was marginal, with some controls unreachable or inoperable at various suit pressures. The suit required 10 to 18 minutes donning time with assistance. Pilots desired increased torso and shoulder mobility; all agreed that the suit impaired mobility and vision. Increased finger dexterity, particularly under pressurization, was also recommended. Hall and Martin (82) testing for effects of prolonged exposure in the Mark III, Mod. II, suit found that the subject could walk and move his arms and hands at full pressurization, but his fine finger movements were restricted.

Burns and others (83) tested the Mark IV suit for operator mobility in the supine position (employing a Mercury Contour Couch.) Measures of reaction time were obtained for four subjects actuating switches within a radius of reach at seven positions from left to right: (1) a rocker, (2) a guided push button, (3) a short stem toggle, (4) a logn stem toggle, (5) a pull throw toggle, (6) a regular push button, and (7) a 10 position rotary. The simulated console position was varied about the

subject's perpendicular plane at arcs A, B, and C, while maintaining reach limits. Arc A was 31 inches above the reference point; arc B was in a two-inch raised position, and arc C lowered two inches from the 31-inch level. Figure 46 presents the RT data plotted in comparison of performance with the insulated Mark IV suit with integral ventilation tubes; unpressurized, and pressurized to 5 psi. Note that the major RT increases seem to be at positions off the straight-away reaching areas of the left or right hands.

Burns and Burdick (84) obtained similar results in a study of the Mercury astronauts. Each astronaut was tested in the Mercury capsule console mockup while wearing his pressure suit with ventilation pressure only, while performance was compared to that under full pressurization or 5 psi. The reaction time measures consisted of measures of time required for the subject's single or serial responses in actuating appropriate switches or controls to any number of indicator light signals from one to twelve. The differences in RT measured for these conditions of suit pressurization were statistically significant. RT for the 5 psi condition was greater than for 0 psi. Indeed, certain controls could not be properly actuated, and at times even reached by the astronaut while wearing the inflated suit. Also, inadvertent actuations were frequently made in the inflated condition involving contact with adjacent toggle switches, some of which were supposedly protected with lucite guards. The investigators conclude that; (1) the effects of learning on performance under a 5 psi condition should be further studied, since evidence suggests that speed and accuracy might be reduced in adaptation to a normal operating range. (2) work place layouts should be well planned for easy reaching and control actuation by the pressure-suited operator, (3) valid measures of performance efficiency must be developed to include energy expenditures as well as motor performance; for example, in this study pulse rate increased from 66 to 80 beats per minute under the 5 psi condition while the operator worked on identical task profiles.

It is of interest to note that the overall mean reaction time is approximately 0.9 seconds for the data presented in Figure 46 with 0 psi pressurization as compared 0.85 seconds for the data presented in Figure 43. Although the specific responses are somewhat different, it does suggest that there is relatively little lost by wearing the uninflated suit. However, when the suit was inflated, the reaction time was considerably greater, the overall mean being approximately 1.6 seconds, or approximately 70 percent greater. If this is extended to comparable manipulation activities in a space suit, they can be expected to take approximately twice the time they would take in "shirt sleeves."

Passman (85) conducted tests in a cockpit simulator to find the effect of a full pressure suit on operation of three types of controls: (1) a bar knob with skirt dial, (2) a fluted MS type knob with horizontal digital display, (3) two pulse buttons with remote indicators. The controls were employed by subjects using their bare hand, with hand in pressure-suit glove simulator at 1/2 psi, and at 3 psi. The subjects

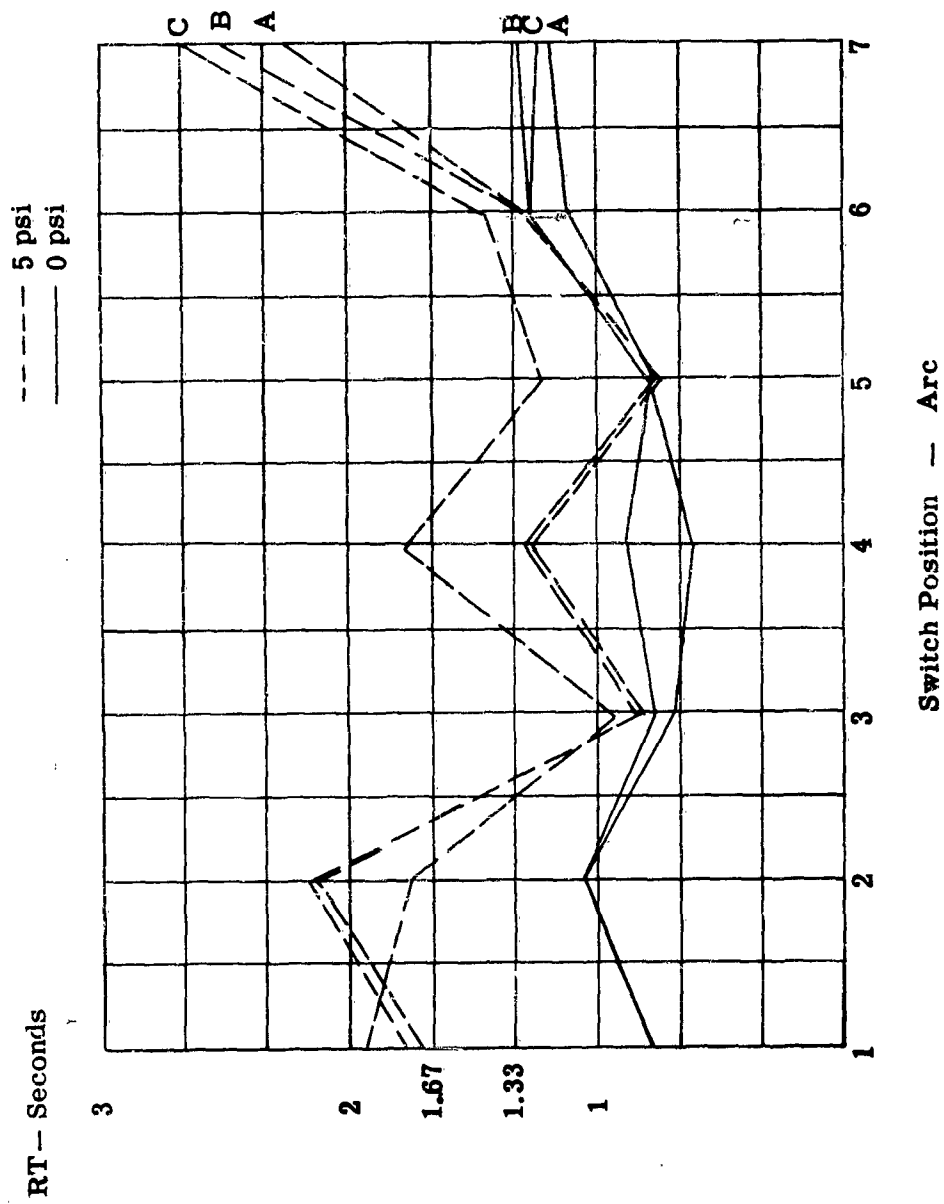


Figure 46. Comparative Switch Positioning Reaction Time in Pressure Suit
(After Burns, 83)

were requested to set up randomly selected numbers between one and twenty with each control, while a measure was taken of set-up time and a rating was made by each subject for each control configuration under the described conditions. Results are presented in Figure 47. Note that for all controls, the ratings tended to become less favorable and set-up time to increase with increasing pressurization. The bar knob with the skirt seemed to give the most trouble, probably due to bulk of the pressurized glove obscuring the view for setting. The fluted knob was rated superior with 1/2 psi due to a high torque and force required tending to irritate the bare hand. In general, however, finger dexterity appeared to deteriorate with increasing pressurization at the hand.

A study recently conducted at Rocketdyne (62) presents some very important data relative to the problems associated with space maintenance conducted in a pressurized suit. In this study two maintenance tasks were attempted on a J2 engine (200,000-pound thrust): Remove and replace an oxidizer bypass duct, and remove and replace a gas generator spark-plug assembly. Tests were conducted on two subjects under conditions of no pressurization and pressurization with a 3.5 psi differential. The major results of this study were:

- (1) Inflated gloves do not enable a firm grip on conventional tools. The best tool diameter appearing to be approximately one inch.
- (2) Inflated gloves do not enable handling small parts (nuts and washers). The subjects for example could not remove 9/16 inch nuts during the tasks.
- (3) Task time was approximately twice as long when the suit was pressurized.
- (4) One glove was cut during a maintenance task.
- (5) After one hour working in the pressurized suit the subjects experienced some nausea and stomach discomfort.
- (6) The suit could not enable complete straightening of the trunk and therefore, caused some discomfort.

The implication of these studies is far reaching relative to the conduct of inspace maintenance. Of critical importance is the possibility of suit puncture during maintenance activities. The data on discharge phenomena presented in Chapter 5 indicated the severity of even a very small puncture.

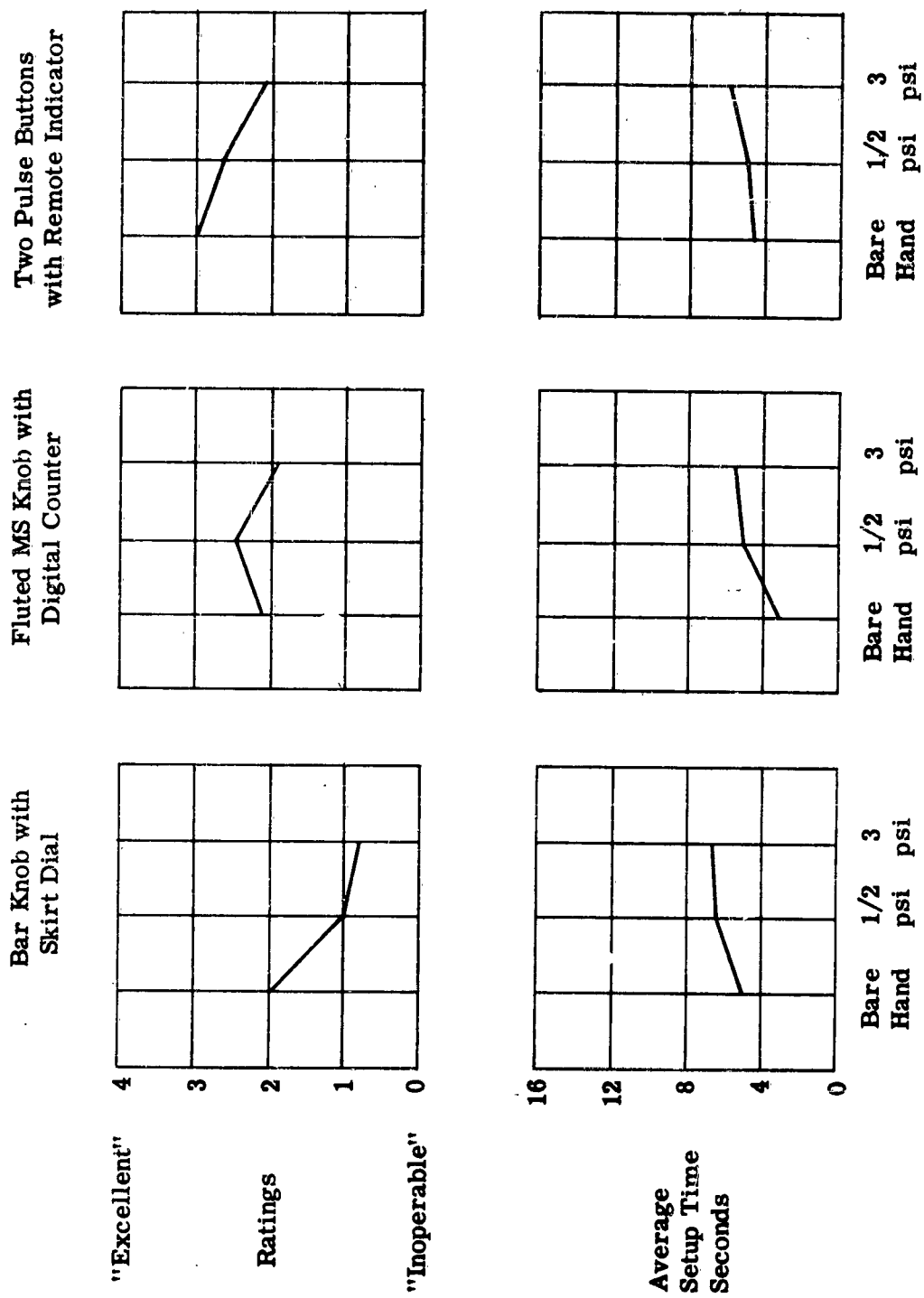


Figure 47. Comparative Manual Dexterity Data with Pressurized Glove (After Passman, Reference 85)

Since maintenance tasks, here on earth result in numerous impacts capable of breaking the worker's skin we must recognize the high probability of similar impacts on sharp objects during space maintenance tasks; especially when limb control is more difficult due to the limb and mobility restrictions and effects of Zero-G.

In summary, the effect of a space suit on a worker's capability to perform maintenance tasks will be to reduce the spectrum and quality of work he can accomplish. This conclusion applies equally well to tasks attempted both inside and outside a space vehicle. Degrading factors such as restrictions in limb and body movements, finger dexterity, longer reaction and performance time, tactical sensitivity, visual field, and special muscle fatigue may be compensated to some extent through training, special design of tools, hardware, workplace layout, and access areas. Until such time that an operationally acceptable space unit is available for testing, specific design recommendations cannot be made. Further, it will not be impossible to predict, with assurance, the times required to accomplish in-space maintenance tasks, in a full pressure suit, until realistic configuration data defining the components to be maintained are available.

E. THE VISUAL ENVIRONMENT

The orbital worker will be confronted with a number of demanding visual and perceptual problems, due to the photometric environment of space and orbital mechanics phenomena, if he exits his vehicle to engage in maintenance or assembly operations. First, he must locate his target object visually if it is not physically coupled to his parent vehicle, and control his transfer trajectory if he releases physical contact with his parent craft. Secondly, he must be able to protect his eyes from the sharp visual contrasts that may be experienced and adjust to the relatively unique visual field of space present at and around his work area.

1. Location of the Target Vehicle

In order to determine the worker's ability to visually locate the target vehicle, it is first necessary to define the size and phase relationships between the observer, the target vehicle and natural light sources for different relative locations. Although the average human can detect objects which subtend one (1) minute of arc or even less under ideal conditions, it is probable that under orbital conditions he could detect only objects which subtend considerably larger angles.

To illustrate the phase relationship problem, consider the example of a satellite orbiting Earth with the repair ship (and worker) in a nearby concentric orbit. Figure 48 demonstrates the phase relationship problems. Neglecting all light sources except the sun, moon and earth, we can define certain distinct cases of different photometric conditions for the maintenance worker.

In Case I, the observer would have difficulty in seeing the target when looking at it with the illuminated Earth as the background. This is so since the illuminated Earth could have a luminance of 10 lamberts or greater, thus constituting a glare source brighter than most sources ever encountered on Earth. In addition, since the target is also illuminated by the sun, and assuming similar reflectances for Earth and target, the contrast between target and Earth background would be low, making detection difficult even if the glare is reduced by goggles or filters. As the target shifts to a space background, still illuminated by the sun, chance of detection is enhanced because of increase in contrast and elimination of the glare background. The effect of the moon for Case I is negligible.

In Case II, it will be of extreme difficulty for the observer to visually locate the target when the sun's image forms the background. Since the sun's luminance is approximately 7×10^5 lamberts any direct observation of such a glare source would blind the observer. Contrast in this case has the maximum negative (dark object against bright background) value of -1. Use of filters to reduce glare might allow visual detection since the contrast should theoretically remain constant. However, any scattering of light in the filter would reduce the absolute value of the contrast. As the target moves out of the range where the sun is in its immediate

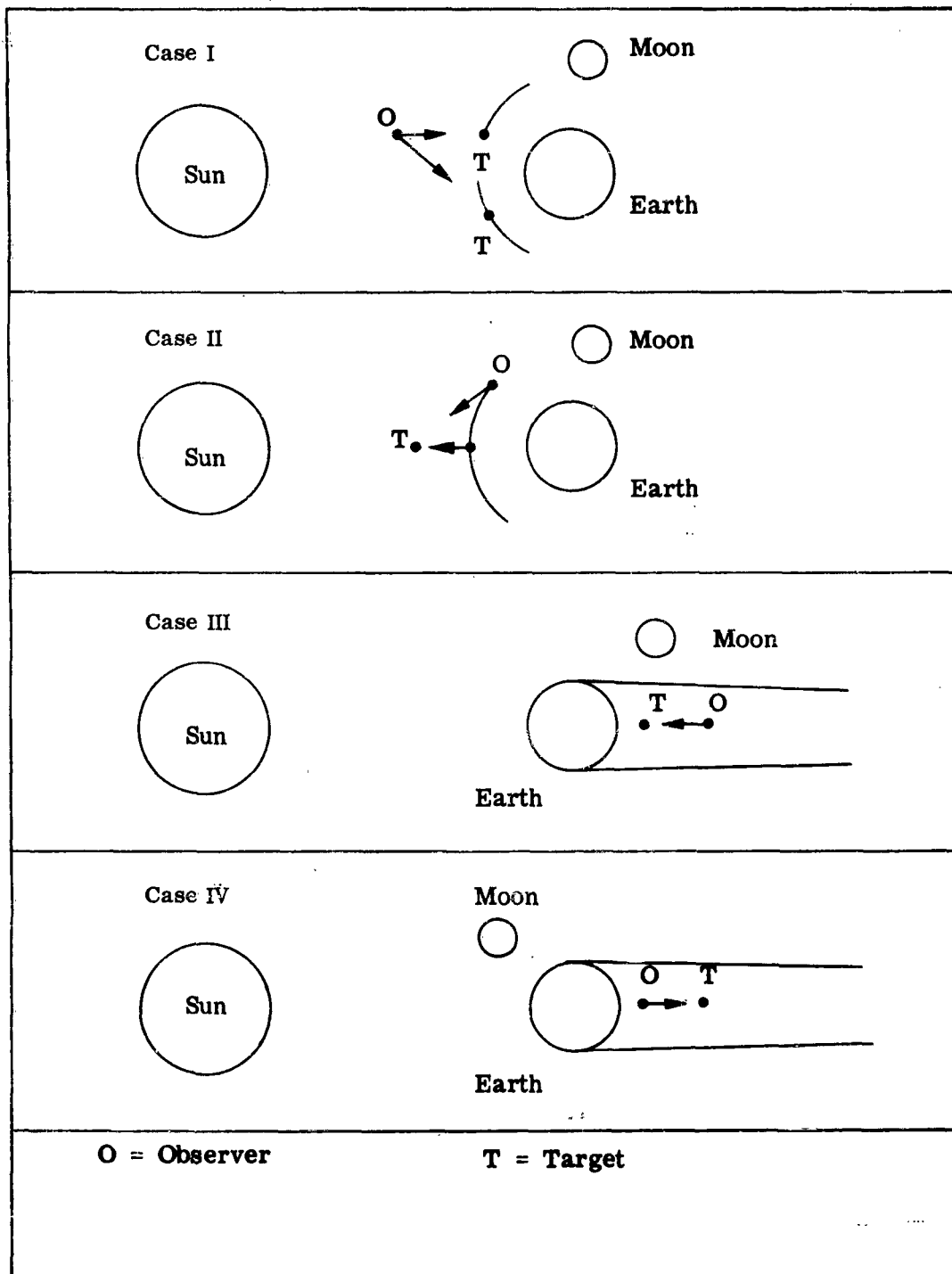


Figure 48 . Orbital Worker - Target Relationship

background, visual detection should be possible if the sun's image is blotted out of the observer's field of view by an appropriate shield. Although only part of the target's face (as seen by the observer) is illuminated by the sun, it should not be difficult to see since it will also be illuminated by the daylight face of the earth. Again here the moon has little influence.

In Case III, both target and observer are in the earth's shadow. If the moon is in proper phase to illuminate the target, it should be possible to see it against the dark Earth background. Without moonlight incident on the target, it will be extremely difficult to locate the target visually against the Earth background without auxiliary lighting.

Case IV is similar to Case III when moonlight is available for target illumination. Chances of seeing the target against the starfield background during dark moon conditions may be feasible if the target subtends a fairly large angle. Since it will be possible to detect 3 to 4 times as many stars with the unaided eye from orbit than on Earth under ideal conditions, the target may blot out a part of the star background and thus be detected as a hole in the observer's star-studded "sky".

One cue which will be an aid to the visual detection of the target vehicle in all cases will be its motion relative to the other celestial bodies. If the observer is stabilized, the target will remain relatively fixed in his field of view but be in motion with respect to other celestial bodies in the field of view since both he and the target approach the same orbit and velocity as rendezvous is achieved.

One phase of this problem of visually locating the target which must be studied in greater detail involves the perception and reaction times as well as the dark or light adaptation times for the worker. The difficult problem of placing a maintenance worker in the same orbit, or a concentric orbit, close to the object to be maintained means that the two will, in general, tend to drift apart unless continuing corrective thrust are applied or copulation is achieved. The time required to dark adapt varies from 5 to 30 minutes depending on the luminance of the object to be observed. Assuming an earth satellite in a 90-minute orbit, approximately 20 percent (18 minutes) of the time will be spent within the earth's shadow. Thus, it may be that the worker waiting to dark-adapt before looking for his target vehicle may not have sufficient time to find it before his "daytime" begins and he must start to light-adapt. Meanwhile the worker may, even with the slightest difference in eccentricity of orbit, drift kilometers away from the target vehicle.

Depth perception, in the space flight environment, will ordinarily be of little value because of the enormous distances involved. During approach and rendezvous, however, the worker may utilize binocular depth perception to approach the target craft if he has previous training and knowledge of its size. Depth perception by monocular successive parallaxes (motion parallax) is feasible at high speeds, but during approach it is generally desirable to minimize closing speeds to avoid impact or overshoot. Depth perception in space without the ordinary visual cues presents a critical problem for the space worker closing on an orbiting vehicle.

A translation scheme which appears feasible for both an extravehicular pressure suit system or a maintenance capsule has been evaluated in a recent Air Force Study (10). In this approach the man aligns himself so that the target object appears to be directly in front of or above him. He thrusts directly toward the object until he reaches a desired relative velocity with respect to the object. He now would coast toward the object at a velocity which is higher than the target object. To compensate for radial motion, because of his higher orbital velocity, he would either realign himself and thrust toward the object again, or fire his vertically oriented thrusters without realigning his body. When, at some later time, he notes that the target object has moved below himself so that the target makes an angle with his straight-ahead reference system, he applies a corrective thrust. Thus he flies an oscillatory flight profile, as is presented in Figure 49, around the orbital altitude of the target object continually decreasing the distance, and finally obtaining contact with the target object. As he approaches the target he retrothrusts so as to make contact with zero relative velocity and avoid damage to his support mechanisms and/or vehicle structures.

2. Visual Adjustment at the Work Area

The space worker, whether engaged at work internal or external to his space vehicle, must possess an adequate visual capability and be provided with adequate illumination of the work area. Visual problems within a stabilized orbiting vehicle should be little different than those of comparable lighting problems in an earth-bound environment. However, visual problems, external to the vehicle, may be considerably more involved. In performing visual inspection checks or in performing visual-motor tasks, the illumination and contrast, and the interacting effects of relative motion, will have a considerable effect on the operator's capability and his lighting requirements. Moreover, the cyclical aspects of vehicular or moon orbit will at times place the operator in relatively complete darkness by occultation of the sun by earth or moon, or by the vehicle itself.

The primary source of illumination for our solar system is the sun. Since the illuminance falls off inversely as the square of the distance from the sun, it is possible to plot illuminance levels for various regions of the solar system. This is plotted in Figure 50 after Strughold and Ritter (86). We have indicated on this graph certain illumination conditions commonly encountered on Earth such as bright sunlight, etc., for comparative purposes. Also, from a knowledge of the reflectances (albedos) of the various celestial bodies, it is possible to compute relative luminances. These data are presented in Table XXXVI. These bodies, when reflecting on the orbiting assembly at which the operator is engaged in maintenance activities, will illuminate it with varying intensity, depending upon the reflectivity of the satellite surface.

The relative shadowing effects that would be encountered when the vehicle is on the dark side of the earth are a function of the vehicle altitude and solar depression angle as well as the vehicle direction in geographical orbit trajectory.

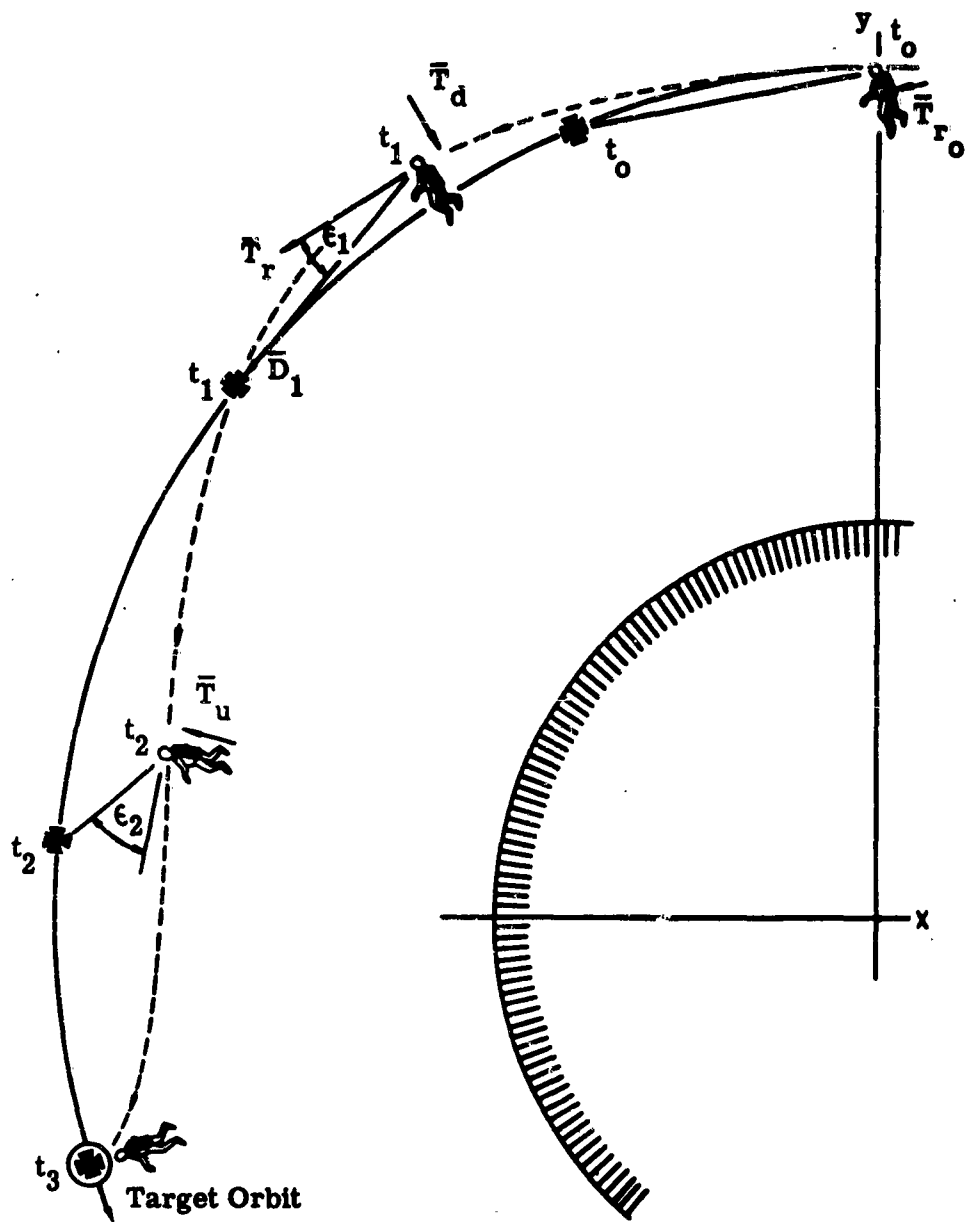


Figure 49. Typical Transfer With a Manually Operated Self-Maneuvering Unit

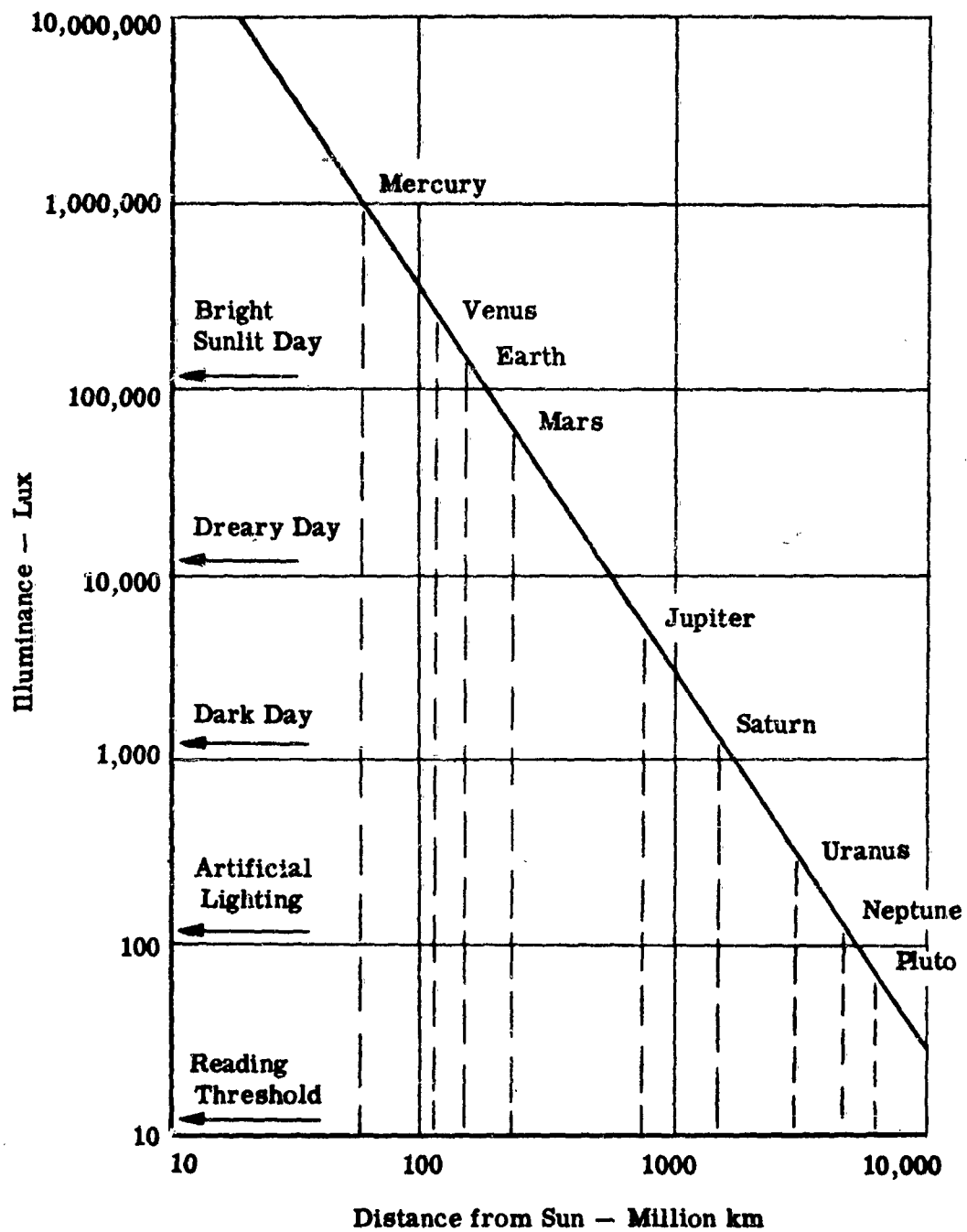


Figure 50. Range of Solar Illuminance

TABLE XXXVI
RELATIVE LUMINANCE VALUES

Astronomical Body	Distance From Sun Km x 10 ⁶	Reflectance	Luminance (Lamberts)
Mercury	58	0.058	6.45
Venus	108	0.76	15.8
Earth (overall)	149	0.39	13.4 (max)
Mars	228	0.148	0.929
Jupiter	778	0.51	2.90
Saturn	1462	0.50	0.964
Uranus	2869	0.66	0.239
Neptune	4495	0.62	0.113
Pluto	5900	0.16	0.00657
Moon		0.072	1.2
Sun			700,000

Whether or not the maintenance worker will require artificial lighting to illuminate his work area depends not only on the illuminance of natural light sources such as the sun, but also the shadows cast across the work area by himself or other objects. Because there is no atmosphere to scatter and diffuse incident light, shadows will be very prominent and of high contrast. If it is necessary or desirable for the worker to perform in the shadow areas, it will be necessary to use artificial lights or have reflectors at appropriate locations to adequately illuminate his work. For example, it has been estimated by Baker and Steedman ⁽⁸⁷⁾ that, while in the shadow of the earth, a satellite lighted by the moon's reflection would have a luminance of 0.01 foot Lambert.

Special design may be required for wide-angle flood lighting, under shadow conditions, to permit use of peripheral vision, since no light diffusion or absorption will occur in space. On the other hand, protective provisions must also be included in design of head or eye pieces to prevent deleterious visual effects from intense stimulation and simultaneous or successive glare.

If we generalize from available laboratory data on vision, man's visual capabilities, with respect to certain aspects of the space environment of the maintenance workers work area is pretty well defined. For example, when his orbiting structure moves into the completely dark shadow field of the earth or moon, he might be expected to dark-adapt and detect various increasing low values of white and colored light against time as plotted in Figure 51.

If the space-maintenance worker were to illuminate his work area, his capacity for seeing fine details of his work, or his acuity, might be expected to occur as a function of the luminance intensity, as plotted in Figure 52. In this regard, Blackwell's task - illumination requirements data, reported in the Illuminating Engineering Society Handbook (1959) might apply. Recommended illumination levels for various maintenance areas, and tasks are presented in Table XXXVII.

If relative motion occurs or is induced between the worker and the maintenance structure on which he is working, he might be expected to perceive detail in the moving parts as plotted in Figure 53.

When the space maintenance worker is exposed to a high luminance source, as with a glossy surface reflecting the sun, or in leaving a vehicle of high intensity illumination, he will be able to see a light source of some lesser intensity than when fully dark adapted, as plotted in Figure 54.

On the other hand, when he is exposed to a lower level luminance, and is stimulated by a relatively more intense luminance; e.g., his work area revolves into an area of high sun reflectance, he may be expected to experience discomfort or pain with respect to relative luminance values as plotted in Figure 55. If he is exposed to even momentary high intensity luminance, approaching the luminance value of the sun, he will lose visual sensitivity for low luminance values in the order of magnitude and recovery time as suggested by the plot in Figure 56. If his work area is a small segment of illumination with a relatively high-intensity wide background, such as the light side of the earth or moon, he will experience simultaneous-glare discomfort as suggested by the plot in Figure 57.

The foregoing data derived in a laboratory setting, however, may have limited application to the space maintenance worker's actual visual situation, since the magnitude of experimental luminance values on which present data are based, is considerably less than that to be encountered in space. Pigg (94) has studied visual acuity under conditions of short-period exposure to weightlessness in Keplerian trajectories. He was concerned about the physiological visual effects that have been found under transverse g condition; i.e., where theoretically the crystalline lens displaces in the direction of gravity. Other possible consequences on vision of a variable g environment were cited, such as autonomic nervous system involvement, or distortions in eyeball shape. Tobias and Slater (95) have discussed such conditions as radiation on vision, and cited the major but indefinable problems of stress

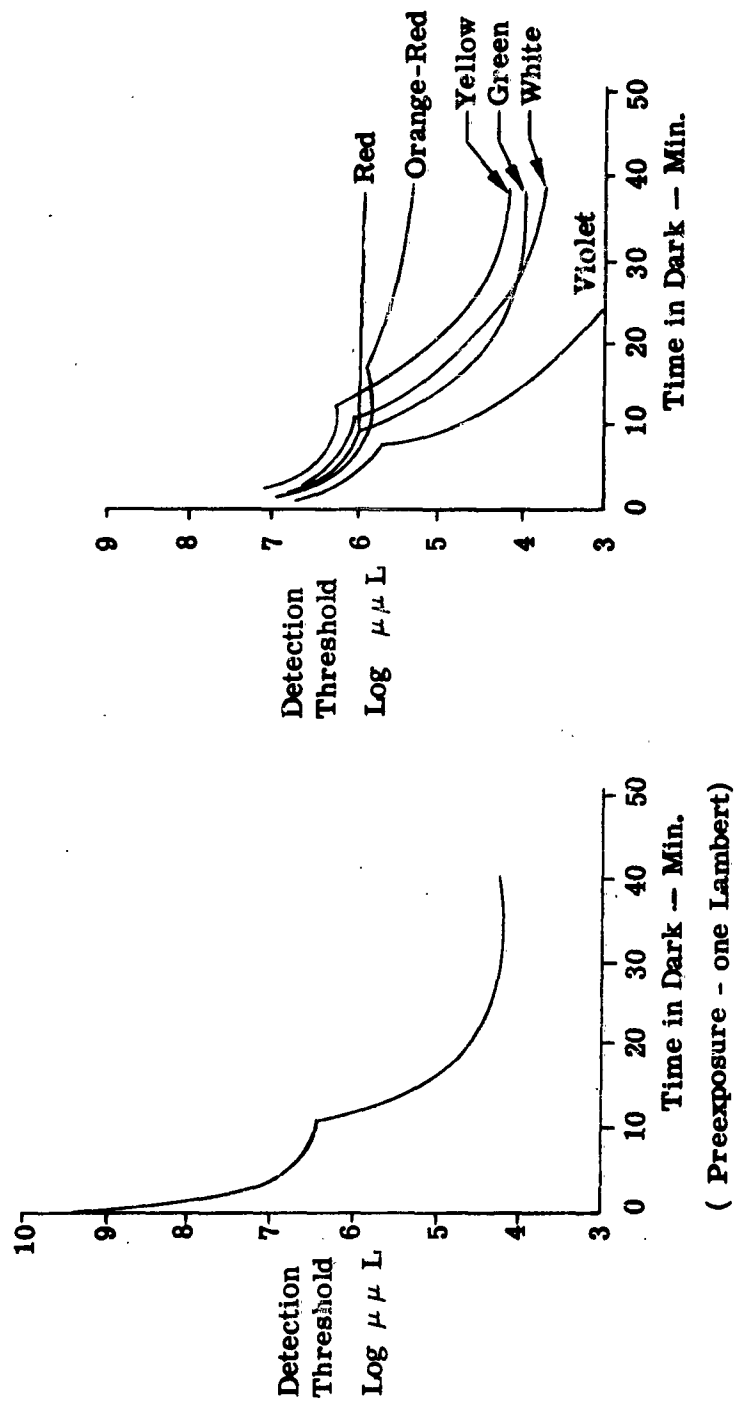


Figure 51. Dark Adaptation (After Wulfeck, 88)

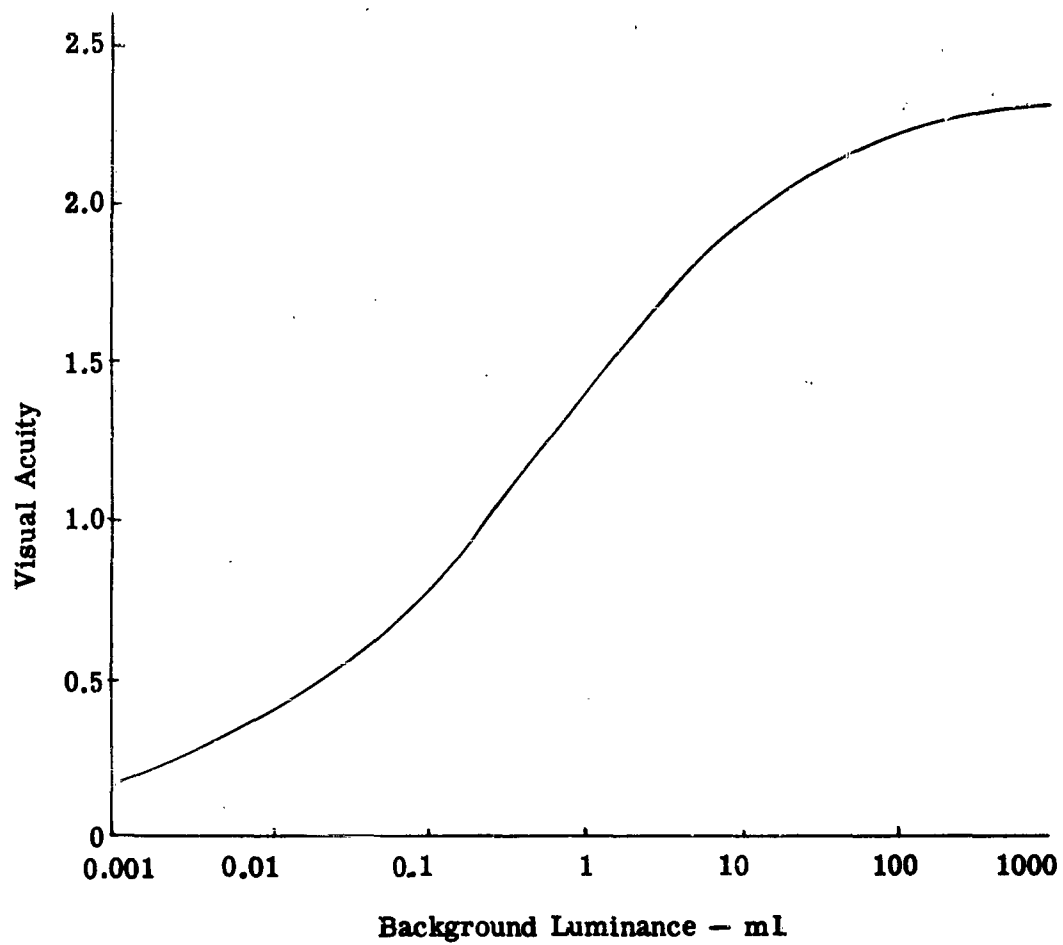


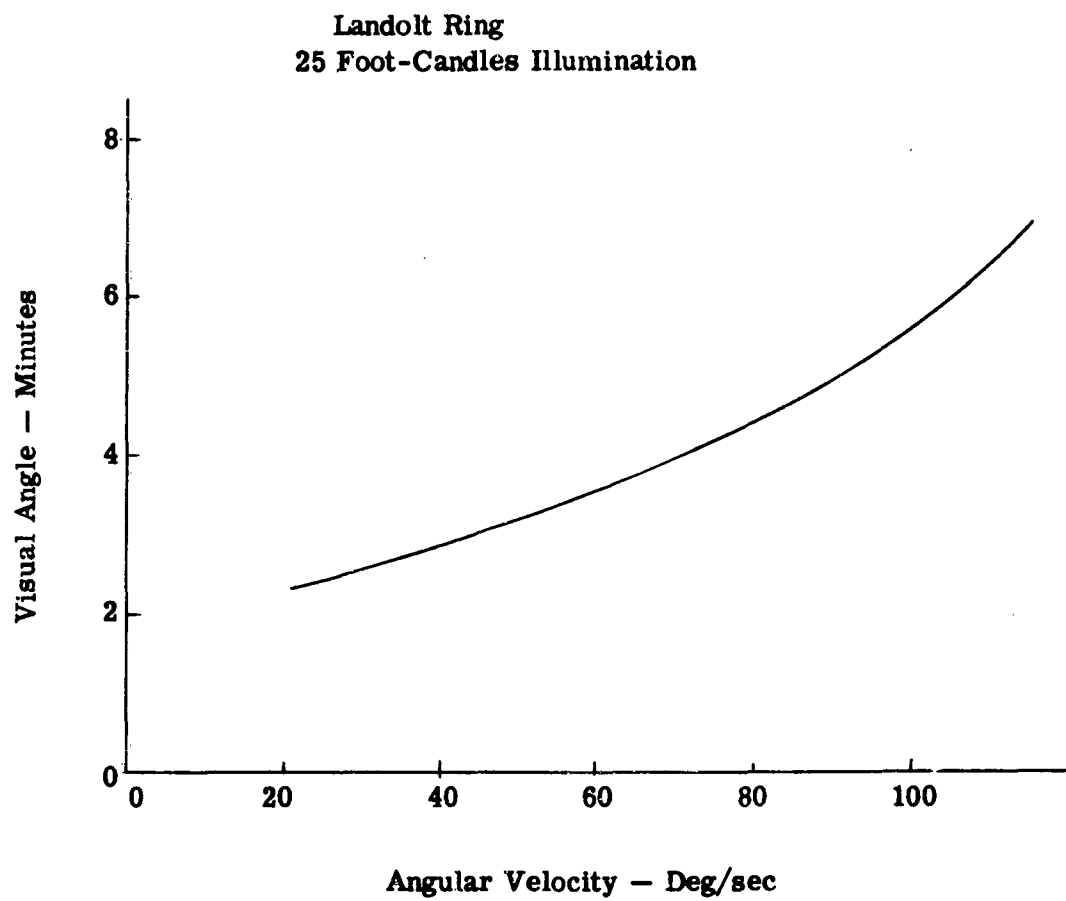
Figure 52. Visual Acuity as a Function of Luminance (After Moon and Spencer, 89)

TABLE XXXVII
RECOMMENDED ILLUMINATION LEVELS (LIGHTING HANDBOOK,
ILLUMINATION ENGINEERING SOCIETY, N. Y. 1959)

Maintenance Task or Area Description	Illumination * - Foot Candles
Drilling, Riveting, and Screw Fastening,	70
Welding: General	50
Supplementary	1,000
Assembly:	
Rough (easy seeing)	30
Rough (difficult seeing)	50
Fine	500
Extra Fine	1,000
Repairs	
Inspection:	
Ordinary	50
Difficult	100
Highly difficult	200
Very difficult	500
Most difficult	1,000
Areas:	
Catwalks	2
Main Entrances	10
Oil Storage Tanks	1
Work Shops	20
- on work	50
Reading and checking:	
Typed original, good ribbon	1
Typed original, extremely poor ribbon	3,140
White chalk on grey	21
Brown spot on red	10,000
Brown spot on tan	10,000
Chipped wood grain (viewed from 30 inches)	1,030
Reading vernier calipers:	
Non-etched	631 **
Reading new micrometer etched	7.4 **
Reading old micrometer	
Specular on numbers	282 **
Specular on divisions	7.6 **

* It is generally assumed that brightness of the peripheral field is uniform and equal to the immediate background of objects to be seen with central vision. All laboratory data are based on this condition; where gross departures from this are indicated, new values must be determined.

** Foot Lamberts



**Figure 53. Visual Acuity with Relative Motion (After
Ludvig and Miller, '90)**

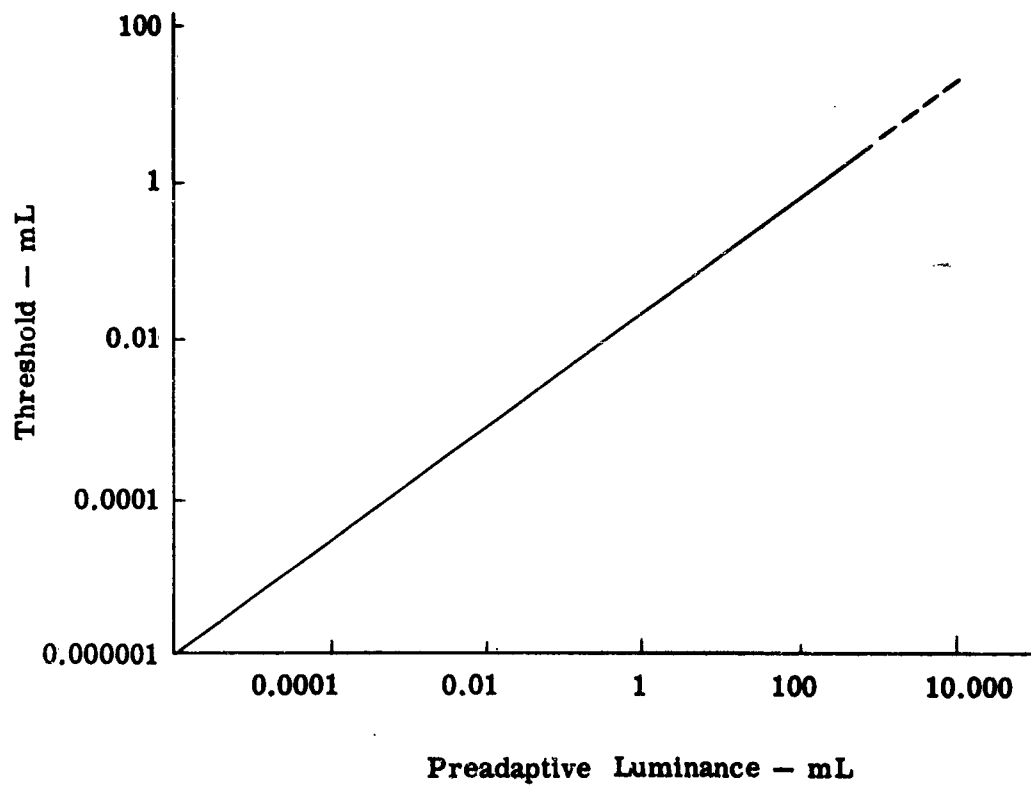


Figure 54. Threshold Sensitivity as a Function of Preadaptive Luminance (After Chapanis, 91)

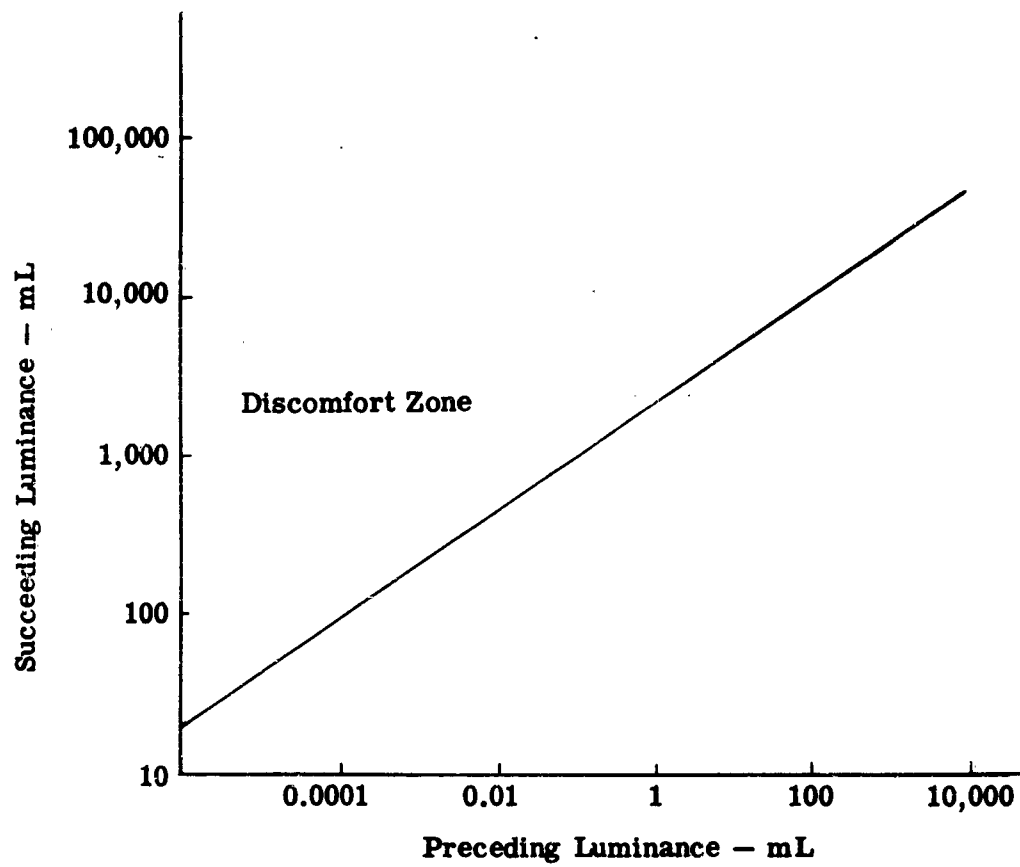


Figure 55. Successive Glare Effects (After Chapanis, 91)

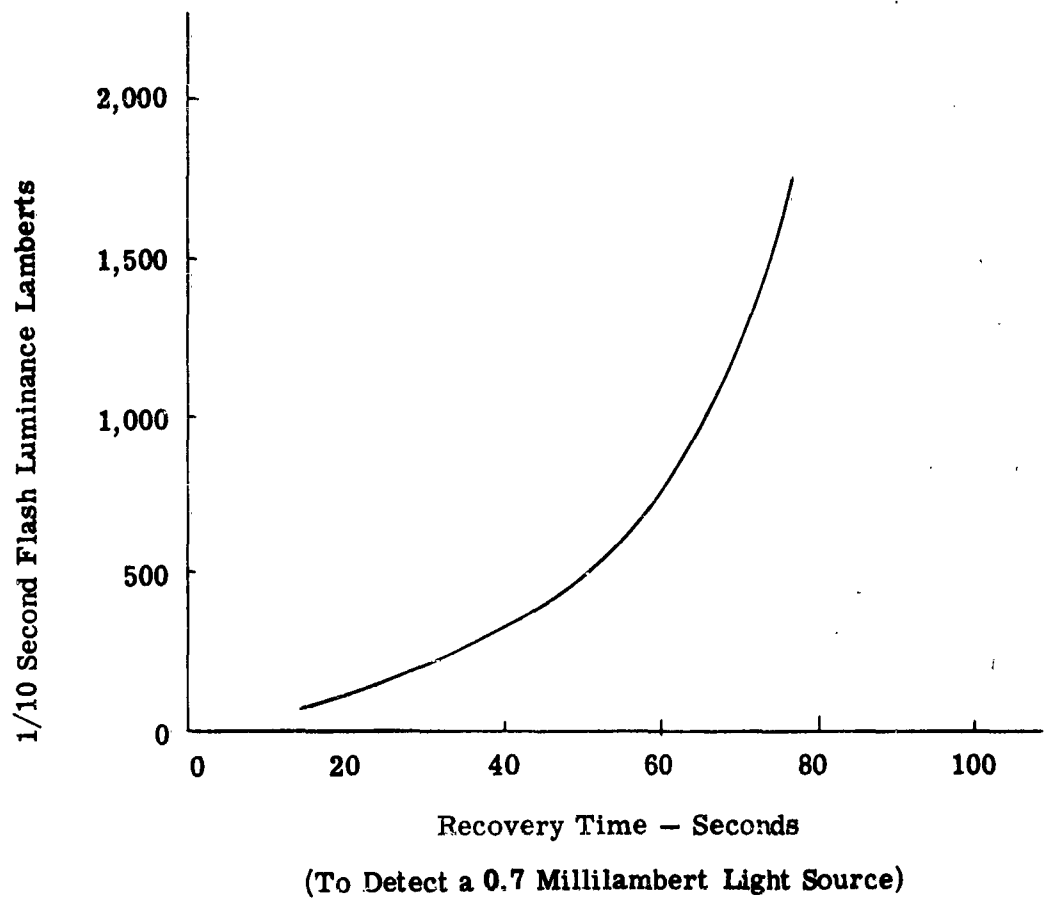


Figure 56. Flash Blindness Recovery Time (After Metcalf,92)

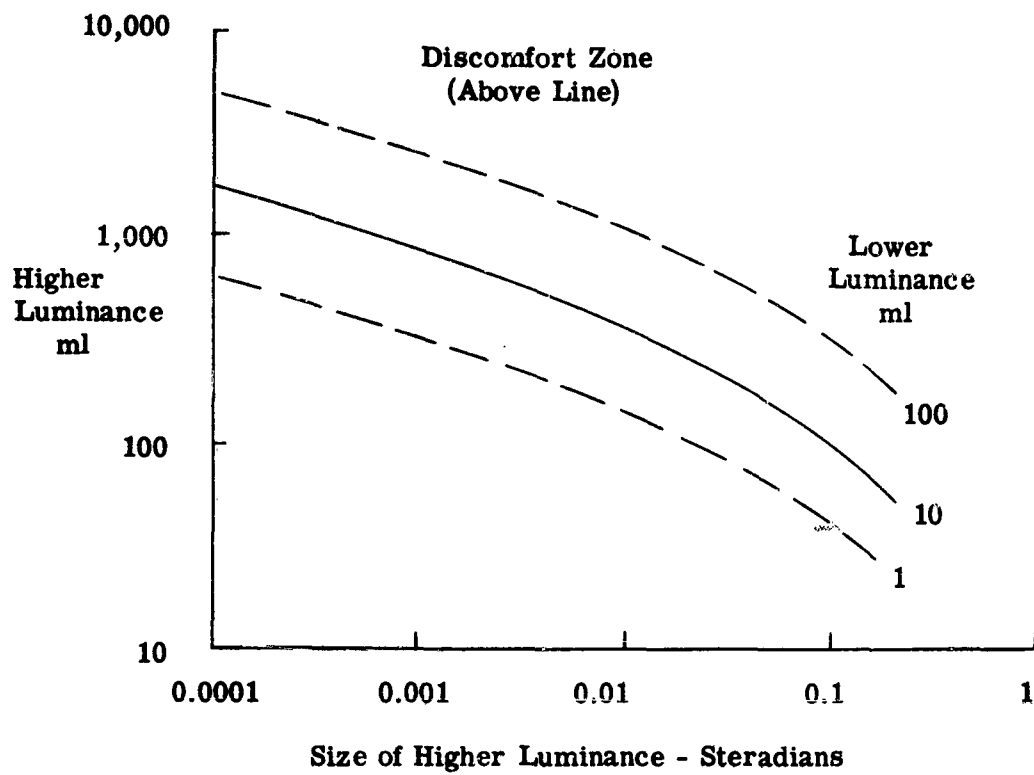


Figure 57. Simultaneous Glare Effects (After Guth, 93)

as being those of combined effects; e.g., prolonged weightlessness, spinning and tumbling, high level luminance, radiation, confinement and isolation, anxiety, etc., on the resultant visual capabilities of the space worker. Principles and "laws" of vision, based on data derived in an earth-bound laboratory setting, and formulated within this theoretical framework, may require some reformulation to be meaningful in a space environment.

Within the vehicle, lighting and glare may be adequately controlled by good designs. Outside the vehicle, however, these conditions become a more serious problem. Also, the worker may induce movement to his work structures, and experience uncommon vertigo effects in the absence of fixed visual reference. As he works in the unique visual environment of space, his visual apparatus may become lethargic and inefficient. Such conditions as these must be identified before determination of the tasks that can be performed proficiently can be made.

Fortunately, an increasing fund of research data is developing on visual problems in space. Feldhaus (96) has discussed several visual problems which he feels the maintenance worker might encounter. These are summarized below:

Space Myopia: As he looks away from his work into space, the space worker's eye will not relax its accommodation. Thus he has no way of knowing if he is focused on infinity or an object a few feet away. Provisions for visual references in the work area would alleviate this problem.

Glare: Three types of glare that may be encountered (1) "blinding" in looking directly at the object, as into the sun (2) "veiling" where the object of work is too brightly illuminated for proper contrast, and (3) "dazzling" where bright light enters the eye, is scattered and cannot be focused on the retina. The effects are varying degrees of retinal irritation resulting in discomfort and lacrymation, and, if prolonged, in nyctalopia or night blindness. Adequate shielding must thus be provided.

Direction of Illumination: In the absence of an atmosphere, all lighting is direct and no scattering occurs. In orbit, the lighting may cast direct-shadow effects of an unusual nature. For example, rivets may appear as surface detents. Adequate supplementary illumination should relieve this problem.

Visual Fatigue: Properly corrected eyes will function properly for two or three hours without appreciable fatigue effects. However, if an inordinate amount of close work is required of the space worker, it may result in forms of asthenopia, such as headache, lacrymation, and burning and itching of the eyes, since he cannot effectively look away. Provisions or routines for eye rest may alleviate this.

Heterophorias: The tendency that normal eyes have in an earth-bound environment to spread apart or dissociate may become more pronounced in space if visual cues are lacking. A constant effort may be required to keep the eyes in line, and prevent diplopia or double images. Provisions for visual references in the work area may alleviate the potential problem.

Uncorrected Vision: Due to his almost completely near environment, any uncorrected hyperopia or astigmatism of the maintenance worker will quickly cause discomfort due to the increased accommodative demands. Selection of maintenance personnel on the basis of perfect vision, or vision which may be reliably corrected, therefore, may be mandatory.

Dark Adaptation: The efficient process of adaptation requires optimal dietary and environmental conditions such as the proper atmosphere to prevent anoxia, infections and or non-infectious diseases.

Anxiety and Fear: Physiological effects occur in chronic or acute apprehension, such as pupillary dilation beyond the normal 3 mm diameter. This could interfere with the space-maintenance worker's adjustment techniques and must be considered. Also, psychosomatic involvement from severe psycho-physical demands could result in hysterical blindness. This problem may be alleviated by careful screening of candidates, and maximal assurance for confidence in system and equipment.

While the occurrence of these visual problems is largely conjecture at the present time they do represent a potential source of interference to the accomplishment of space maintenance and assembly tasks.

3. Effects of Weightlessness in the Visual Sense

Pigg (94), using Checkerboard and Schnellen targets, conducted a comparative study of visual acuity in the laboratory, in normal one g flight, and in Keplerian weightless-inducing trajectories. The results did indicate a loss in visual acuity under weightless conditions of six percent compared to normal flight, and ten percent compared to the laboratory environment. However, he considers this loss to be of no practical significance. William and Slater (97) note that when the macula lies in the position of minimum stimulation with the otoliths above it, some nerves are still found to transmit. They suggest that this may be due to normal weight of the otoliths. However, it could also be due to nerve tonus or spontaneity. If the latter held, then the reduced gravity states would not necessarily produce sensations different from those experienced to normal terrestrial accelerative loads.

King (98), however, suggests that the pattern of sensory stimulation in the weightless state will be modified for all modalities but the eyes. In a study of the utricular otolith apparatus of birds in a weightless state (the sensory receptor for eliciting static reflex compensatory poses) he concluded that it did not function under zero gravity even with vision.

Graybiel (99) discussed the likely conditions of spatial disorientation in visual perception, attributable to the physical stimulus as well as the influence of other sense organs. Problems in spatial disorientation in general he relates to celestial darkness and brightness of the sun, inadequate visual framework, vehicle window size, glare effects, stimulation of the semi-circular canals and g receptors, and learning/adaptation to these.

Morant's work (100) has suggested that should the space worker be subject to spinning or rotation about his vertical axis, his perceived medial plane will be displaced from the actual in the direction of rotation. Results of laboratory studies have been interpreted to mean that body tonus, rather than retinal effects, affects the visual perception of space.

Clark and Hardy (101) consider that orientation in space is established by the vestibular system, vision, and mechanoreceptors. Vision and touch, he considers most important for effective reaction. Basic causes of visual disorientation, he cites as linear and angular acceleration effects on otolith organs and semi circular canals. Graybiel and Clark (102) in a study of the stabilizing effects of viewing a complex visual field during rotation had subjects observe black horizontal lines during rotation for two minutes. The lights were then turned out and they were required to set a rotatable luminous line to the horizontal position. Results indicated that deviation of the perceived horizontal following lights out, (e.g., the space worker looks into the darkness) increased fairly rapidly up to about 60 seconds, and slowly for an equal time thereafter. Results also show a gradual though not a simple continuous change from visual to gravitational cues, contrasting with a rapid change from gravitational to visual cues; this suggests the importance of adequate visual cues to the space maintenance worker. In a later study, Graybiel and Clark (103) emphasized the importance of gravitational cues in maintaining vertical and horizontal reference. In this case, they studied the effects of head position on "visual egocentric localization" or the setting of lines in horizontal and vertical planes with respect to gravitational cues. Under static conditions, localization with the head upright was quite accurate, with head inverted, it was inaccurate, and on the side, it was grossly in error. The oculogravic illusion; i.e., apparent visual motion of objects in the visual field, was perceived under centripetal force with the head erect, not exhibited with the head inverted, and not measurable with the head on the side. The authors concluded that interaction of the non-acoustic labyrinth, touch, pressure, and kinesthesia in a gravitational environment and effects of fatigue or adaptation, need to be determined for their effects of the "egocentric visual localization."

Gerathewohl and others (104) have noted from their zero gravity studies that eye-hand coordination was moderately disturbed in the subgravity stage of parabolic flight. However, they consider that the main problem of weightlessness may be that of adaptation or compensation by means of visual and tactile references.

Lansberg (105) suggests that in the weightless state, a space maintenance worker would have to use the visual modality to find the whereabouts of his arms and legs, though some traction would be operative in the interplay between flexor and extensor muscles.

Within the space vehicle, some measure of gravitational force will likely be operating as a function of linear or angular acceleration. For example, Lansberg (105) discusses Van Braun's rotation concept. This is a wheel model with a radius of 40 m, rotating with an angular velocity of $2/7$ radian/sec, applying a centrifugal acceleration (ωr^2) of $1/3$ g. Under these conditions, objects that the maintenance worker dropped at his work station would pursue a path determined by the tangential velocity. Under these conditions, visual, graviceptive, and semicircular sensory information would be conflicting. Head rotations, for example, not parallel with the axis of the satellite would provoke false semicircular impulses. Furthermore, spinning the station would make external tasks even more difficult, since the forces set up would be equivalent to negative g forces. Greening (106) discusses the magnitude and direction of forces on the human body associated with coriolis effects as a function of vehicle spin rates, bodily movement rates, and the orientation of vehicle work areas. Gray and others (107) studied subjects subjected to one and three g's in a centrifuge with angular accelerations of 0.8 and 1.3 radians per second, and while simultaneously rotated around an axis close to the body. There were individual differences, but in general, subjects reported visual coriolis illusions in accordance with predictions of torque generated in the semicircular canals by coriolis accelerations. Guedry and others (108) describe a study in which vestibular stimulation was administered to subjects under differential illumination. Nystagmus was suppressed under illumination, but resumed when the room was darkened. They concluded that a visual fixation field serves to control nystagmus, and both nystagmus and associated subjective reactions tend to diminish in adaptation. Graybiel and others (109) studied the oculogyral illusion in subjects living in a continuously rotating room over a 64-hour period. Sensitivity to the illusion varied among subjects, one perceiving it only when the room was rotated at 10 rpm. The major decline in the illusion and associated disturbances occurred within the first 16 hours.

Coriolis effects on the space worker's in-vehicle visual performance can best be eliminated by minimizing the physical stimulation of the vestibular modality per se; however, the evidence does seem to suggest that effects can be minimized by proper illumination, as well as by a period of adaptation.

Confinement within the close quarters of the space vehicle, and in cases where perceptual isolation may occur, may also impair the space-maintenance worker's visual efficiency although consideration of this overall stimulus complex makes this seem unlikely.

Data reported in the Netherlands (110) on the problem of visual acuity in empty space, indicated that the subjects had difficulty in focusing in empty space,

analogous to disorientation in total darkness. The results showed wide individual differences in the ability to focus under these conditions, with some having difficulty in an empty field of only 20 degrees. The authors concluded that perception of the finer details of an object in empty space is highly improbable.

Another visual problem of the space-maintenance worker may be what Wendt (111) refers to as visual drift or imbalance. This normally occurs spontaneously when a subject is seated head forward in the dark, or with eyes closed. The space worker's work area, as a strong visual field, will most likely inhibit such weak nystagmus, but the evidence suggests that if he looks away such visual imbalance will be established in less than a second, and be suppressed again only in about 1-1/2 seconds after he resumes his visual task component at the work area.

Although a detailed analysis of the perceptual aspects of all of the maintenance task was not accomplished, a sufficient amount was done to permit a rough evaluation of visual characteristics. Internal vision should present almost no problems. Ambient illumination can be easily controlled; the visual field contains a large amount of visual detail, etc. It is only when one leaves the confines of the vehicle that vision becomes a serious problem.

In the analysis of the task requirements, one consistent requirement was that for relative fine discriminations. It will be recalled that in the removal of the gas generator injector gasket, 14 small nuts and bolts had to be removed. Normally, one could rely upon tactile sensitivity as an aid in removing the bolts. In the pressure suit, with its attendant reduction in the pressure sensitivity, more reliance must be placed on the visual sense. Each of the extravehicular tasks analyzed required fine discrimination for aligning, inspecting, removal/replacement of parts, etc.

Precise and detailed prediction of the space maintenance worker's visual capabilities at the work area must await further data. Jones (112) has discussed significant conditions in visual capability to be adaptation, ratio or interior/exterior, ambient illumination and object background contrast. He concludes that definitive information can only be obtained from manned orbital missions, where adequate observations and recording is possible.

The preceding review and discussion of the perceptual environment of space indicates the degree of uniqueness to be anticipated. Although the present data do not indicate any severe or debilitating factors that would prevent a space worker from functioning within useful limits, there is little doubt that the space worker must be prepared to develop a new set of expectancies and new protective and augmentation accessories must be designed.

F. WEIGHTLESSNESS

An increasing volume of literature has appeared in the past decade on the effects of the weightless condition that will be typically encountered in orbital flight. Brannan (112) has reviewed the problems of muscular deterioration, circulatory changes, motion sickness, and movement coordination attendant upon the weightless condition. Burch and Gerathewohl (113) conclude that the stresses in a weight free state are well within the range of tolerance of the human and animal organism, although there is a tendency toward prolonged and fluctuating tachycardia in the early stages, and decreased cardiac activity in the later stages of weightlessness.

Momonacá (114) found that displacement of the thoracoabdominal viscera, as well as deficient motor coordination, occurred in a simulated weightless condition, although this may be, in part, a function of the simulation.

Belles (115) has reported that cosmonaut Titov was motion sick during most of his 17-1/2 orbit flight. The condition seemed to worsen when he turned his head sharply, and eased somewhat after a nap. He did, however, maintain a sufficient level of working capacity at all times. Clark and others (116) have concluded that the organism has much capacity to adapt, though man's ultimate adaptive capacities are unknown. Also, his toleration and cross-adaptation to conditions of combined stresses are unknown.

Though the effects of extended periods of weightlessness on the space maintenance worker are not thoroughly understood, the majority opinion among experts seems to be that adaptation in the physiological reaction, and training in the coordination functions together with provisions for work attachment and other job aids, should provide for effective job performance. The effects of weightlessness on task performance is discussed in detail in the Motion and Manual Dexterity and Extravehicular Suit Restrictions Sections of this Chapter and Chapter 8, Tools and Fasteners.

G. DISCUSSION AND RECOMMENDATIONS

The present state-of-the-art in space flight leaves many unknowns confronting the space maintenance worker, however, early experiences such as the Mercury flights, have indicated that man can fulfill many requirements in his observations and flexibility of behaviors that cannot be designed into self-contained automated space systems. Thus the one major claim for the employment of a space maintenance worker lies in his superior adaptability over almost any collection of machines; though to incorporate a man in the system makes it mandatory that careful analyses be made of all parameters influencing his performance, and that design details be worked out to insure safe and proficient execution of tasks.

The following tentative conclusions have been drawn, based upon the study of human capabilities conducted during the contact.

1. Workplace Provisions

Workplace layout for the space maintenance worker, either within or outside the vehicle, must as much as possible permit efficient and safe operation. The following general design considerations should be included:

- (1) Provide for worker tiedown to the work area in a manner that enables him to remain in a fixed position when applying torques.
- (2) Provide access areas large enough for a pressurized suited operator of the 95th centile in a worker population (for critical or emergency maintenance tasks inside the vehicle, or in any area outside).
- (3) Optimize time and motion sequences where emergency repair is necessary.
- (4) Optimize torqueing requirements within the work area, considering the plane of actuation, the traction conditions of the body members, and the support available.
- (5) Establish in detail the advantages of performing a given task, for a specifically required maintenance item, (1) with a pressurized space suit, (2) in a capsule, (3) by means of remote handling equipment.
- (6) Establish feasibility of each specific task allocated to the maintenance worker by means of mockup in simulated environmental conditions.
- (7) Utilize as few different components in the system as possible combining them in different ways to attain system goals and establish a spares inventory based upon malfunction probabilities.
- (8) Establish optimal size of maintenance teams for each specific maintenance item.

2. Provisions for Proper Lighting

Lighting conditions inside the vehicle should be provided in keeping with the types of work to be performed. Recommended illumination levels for various types of maintenance tasks are presented in Table XXXVI. For extravehicular tasks the following should be considered:

- (1) Provisions for shutters, glare shields, and filters to protect the worker's eyes from direct solar radiation and reflected light.
- (2) Provisions for illumination of the work area outside the vehicle. This may take the form of flood lighting emanating from broad area lamps with variac control, attached directly to the vehicles. A design of head lamps, such as those employed by mine workers, may also be considered.

Wide area illumination coverage of the lamp would be necessary in order to permit use of peripheral vision; otherwise the illuminated area would present an extreme contrast with the dark field in the absence of light diffusion. Even in sunlight artificial lighting may be required, since the sun may require shading to prevent veiling glare on the work.

- (3) Provisions in routine for eye rest/work cycles and eye examinations to prevent/detect fatigue and possible eye damage.
- (4) Provisions for selection and screening of maintenance workers for hyperopia and astigmatia.
- (5) Dietary provisions and proper gaseous environment conducive to dark adaptation where necessary.

3. Provisions for the Weightless Environment

- (1) Fastening methods should include "closed-force" systems, such as squeeze riveting.
- (2) Provide harnessing straps. If working on a thin edge, rollers should be attached at the end of the straps; otherwise, lugs or slots should be used.
- (3) Tools should be fastened to the worker in such a manner as not to pick up accelerations imparted in torque or motion reaction.
- (4) Provide locomotion means appropriate to the task of worker translocation or moving equipment in the form of a tug.
- (5) Provide captive stowage provisions; e.g., a magnetized fan, for holding parts such as bolts removed for access.

CHAPTER 7

SPACE MAINTENANCE CONCEPTS

A. GENERAL

Conceived in its full scope, the successful accomplishment of maintenance of any vehicle system requires the development of a complete maintenance system. Such a maintenance system may be considered to include at least the following major elements:

- (1) Logistics — parts provisioning
- (2) Personnel — trained and capable of performing the required maintenance tasks.
- (3) Support Equipments — the tools and equipments whereby successful maintenance may be accomplished.
- (4) Transportation — the means whereby the personnel, parts and support equipments are transported to the object to be repaired and vice versa.
- (5) Maintenance procedures — the techniques whereby the maintenance activities are approached and the system(s) is returned to required performance.
- (6) Maintenance Handbooks — the detailed description of system function(s) and the specification of the procedures whereby the functions can be restored.

In addition to these major elements of the complete maintenance system, it should be noted that when the system(s) is optimally designed for reliability and maintainability that the probability of system malfunction is decreased, the ease with which maintenance tasks can be accomplished is increased, the time required to accomplish maintenance is decreased and, therefore, the complexity of the complete maintenance system is reduced.

In a very broad sense, the development of a complete maintenance system concept includes assembly of the vehicle system at the launch area, pre-launch system checks, postlaunch system and subsystem checks, preventative maintenance at all levels and, finally, malfunction diagnosis and unscheduled maintenance on both ground and airborne equipment. Since the purpose of maintenance planning is to assure the highest possible likelihood of mission success, it is mandatory that the development of a system maintenance concept include all of the aforementioned aspects of the total system. However, the present study, by definition, concentrated on the inflight maintenance requirements; hence, little information has been generated relative to preflight and postflight maintenance requirements.

In agreement with the definition forwarded by Demaree (117), in-flight maintenance is defined as including any activity on the part of a space system component, electromechanical or human, which is directed at restoring the system to a satisfactory operating level. Thus the detection of a system malfunction by a crew member and the manual switching to another mode of operation is as much an example of in-flight maintenance as the actual replacement of the malfunctioning unit. Similarly the automatic switching by a computer from a malfunctioning unit to another mode is considered to represent in-space maintenance. In an analogous sense the docking and assembling of portions of a space station are examples of in-space maintenance insofar as the components, uncoupled, do not represent a functioning system; whereas when joined and functionally mated, they represent such a system.

Current aircraft systems consider "inflight" maintenance only in a cursory sense. This maintenance consists of the monitoring of system performance indicators, observance of signal and warning devices, and perception of out-of-the normal operation (i.e., the sounds connected with lowering of the landing gear, the deceleration force experienced when flaps are lowered or brakes applied, the visual indication of a malfunction). During emergencies, of course, many unscheduled maintenance tasks are conceivable. For example, repairs to damaged control cables, hydraulic lines, electrical cordages, replacement of plug-in components, manual lowering of the landing gear after a power system failure, switch over to auxiliary or emergency electrical power systems, and activation of fire extinguishing networks have all been accomplished during atmospheric flight missions.

Such "inflight" maintenance, however, is only infrequently attempted in current aircraft systems. Typically if an inflight malfunction occurs the crew leaves the vehicle in flight, if the malfunction bears a critical relation to their personal safety, or they operate the vehicle in spite of the malfunctioning system, attempt to land it, and then initiate corrective action by well established static ground based maintenance procedures. The maintenance concept appropriate to aircraft employed for atmospheric missions may be said to include:

- (1) Regularly scheduled ground inspections, calibrations and tests of time sensitive components (i.e., tires, visible fluid leaks, damaged structure, missing components) in the attempt to reduce inflight malfunction probability.
- (2) Periodic disassembly, inspection and replacement of functionally critical components (i.e., engines, tires, electrical and hydraulic power supplies, oxygen supplies).
- (3) Nonscheduled repairs, parts replacement, and troubleshooting (i.e., repairing damaged structure, replacing malfunctioning electrical parts, correcting fluid leakages and repairing tire damages).

Aircraft maintenance concepts have been developed and refined to a point where aircraft, ground equipment, and facilities are designed and integrated for economical and rapid fulfillment of the required maintenance tasks.

With the advent of guided missiles into more common usage, maintenance concepts required some major revisions. No direct manned intervention into the correction of inflight malfunctions was possible; everything had to be accomplished either remotely or automatically inflight, or prior to launch. In this new technology, a different trend in maintenance concepts is thus apparent. As in the conventional aircraft approach, most of the maintenance tasks required by the missiles are still accomplished on the ground prior to launch. But, without a manned crew, any inflight requirements which develop must be accomplished using indirect methods. It is also not necessary to conduct any postflight maintenance activities since such missiles are one-shot vehicles, with few recoveries.

Preflight missile maintenance activities are now directly related to the time element. Inspections, repairs, tests, component replacement, calibrations, servicing, arming, etc., are all time sensitive. During factory checkouts and predelivery inspections, a fairly leisurely pace and relatively detailed tasks are permissible. But, as the missile travels through the various programmed levels leading to emplacement on a launch pad, these activities become broader in scope and require more rapid accomplishment. As the countdown to launch progresses, it becomes necessary to incorporate automated test, service and monitoring functions because of extreme time pressures. Fueling is controlled precisely as to rate and quantity, umbilical connections are removed at a predetermined point in the sequence and malfunctioning or out-of-tolerance components show up as no-go signals on a blockhouse console. However, manned intervention is still required to produce a successful mission. Gantry towers are provided for missile access to permit installation, test, and repair of components. Removal of failsafe arming plugs, visual inspection for leakages and other deficiencies, are accomplished by technicians using tools and test gear. However, once the fire signal is activated, all ensuing maintenance must accrue remotely or indirectly.

The concepts appropriate to this category of system maintenance consist of:

- (1) Deliver component overdesigns.
- (2) Application of redundancy to critical circuits with automatic switchover in case of malfunction.
- (3) Improved reliability of critical components.
- (4) Automated predictive maintenance of electronic equipment.
- (5) Abandon the vehicle (destruct).
- (6) Accept and compensate for degraded performance.

With the advent of manned space flights, it became possible, again, to consider inflight maintenance both inside and outside the space vehicle. The Friendship 7 Spacecraft, though admittedly the first of its type, provides some insight into the space maintenance problems which may be appropriate for such missions.

Friendship 7 Spacecraft spent slightly less than 6 months at Cape Canaveral before it was launched (166 working days). (118) During this time, the vehicle underwent detailed system-by-system tests and also incorporation of the latest design changes. Forty-three days were spent on the launch pad during which 10 days were spent in troubleshooting, 13 days in pad testing, 13 days in system modification because of a faulty fueling system, and 7 days of weather delays. Some of the maintenance activities occurring during this period included:

- (1) Launch complex checkout — Verifies all complex wiring using automatic continuity checking equipment.
- (2) Assembly of spacecraft to booster — This involves mechanical and electrical mating.
- (3) Systems test of the spacecraft.
- (4) Composite spacecraft, booster and facility tests — This included the RF Atlantic Missile Range Linkup.
- (5) Launch simulation — This test validated the spacecraft and booster launch configuration.
- (6) Flight test simulation — A complete check of all spacecraft systems from liftoff to landing.
- (7) Spacecraft servicing — Filling O₂ bottles, tape recorders, landing bag release system, etc.
- (8) Launch countdown.

It is significant to note that during spacecraft servicing a leak was detected visually in the booster fuel tank. Also during installation of the entrance hatch, a bolt was broken. Both these items were repaired on the launch pad by project technicians working from the gantry tower.

During the orbital portions of the Mercury mission, a major task of the astronaut was to observe and report on system operation and to provide control inputs for correction of malfunctions. Secondary roles consisted of obtaining information on visibility conditions, obtaining photographic records and providing physiological information.

It is evident, from transcripts of this first orbital mission, that the most significant in-space maintenance task consisted of monitoring the occurrence of critical spacecraft sequential events and providing corrective manual override functions wherever necessary. Cues to the occurrence of possible malfunctions consisted

essentially of visual observation, auditory evidence, and/or corresponding acceleration from the event itself.

Within the context of The Study of Space Maintenance Techniques, space vehicles of the 1965-70 time period formed the basis of investigation. The representative systems selected ran from a space suited man to a complex multimanned space station having an expected orbital life of five years. From a detailed analysis of the major subsystems making up these advance space vehicles, integration of the results of human capabilities studies and environmental considerations several maintenance concepts have evolved. Before discussing these concepts, however, it appears appropriate to briefly present an alternative point of view, namely that if component reliabilities are continually improved the in-space maintenance approach may not be necessary. Without exception, however, detailed studies of space system reliability have indicated that component improvement alone will not result in space systems with the high reliabilities required for long mission durations. As McRuer, Askenas, and Krendel conclude (119): ".....invention of more reliable devices results in relatively gradual and modest improvements in overall system reliability. The problem is such, however, that orders of magnitude in improvement are needed, and these require breakthroughs in various aspects of the entire technology." (119) If the validity of this conclusion is accepted then it becomes necessary to develop the techniques of maintaining the systems in space, or replacing the systems with comparable ones upon failure. Thus, maintenance systems must be developed which enable the restoration of the system to a performance level compatible with its mission requirements once a malfunction occurs. It may thus be concluded that manned systems require either a system maintenance capability or the presence of another system for recovering the crew when disabling system malfunctions occur.

Before discussing various approaches to accomplishing in-space maintenance, and the requirements that they impose upon present technology, it becomes desirable to define in general terms the difference between in-space maintenance activities and the maintenance activities required of earth-based systems. Consider the problem of space maintenance within the context of system reliability, for it is the presence of system components with reliabilities of less than 1.00 which results in the primary requirement for maintenance activities. The conventional definition of reliability which has been used with ground-based and atmospheric flight systems has been based upon the criterion of mean time to failure. With atmospheric flight systems, as long as no attempt was made to accomplish inflight maintenance, the mean time to failure criterion of system reliability appears to be appropriate. However, once the requirement is imposed upon the system for inflight maintenance, a second measure of system reliability should be introduced. This measure may be called mean time of restoring the system to a satisfactory operating level. With conventional aircraft systems, if a flight malfunction occurred, the crew left the vehicle if the malfunction bore a critical relation to their personal safety, or they operated the system with the malfunction, landed it and then the activities concerning restoration of the system to its former operating level were initiated.

Thus, except for unscheduled emergency tasks which could be handled with relative ease during flight, the nature of mean time to restoration of the system performance was critical only from the secondary point of view of restoring the system so that it could operate again at some later time. The restoration time should be minimized according to the use requirements for the system. With space vehicles, however, almost any system malfunction affects directly: (1) the immediate and long term safety of the flight crew; (2) the capability of the system to reenter the atmosphere and to be recovered; and (3) the loss of a system whose on-station requirements may be critical for military or scientific purposes. Thus, with space vehicles, the primary concern from a maintenance point of view is to restore the system to an operating level while the system is on station and within a time period determined by the mission requirements and crew safety.

Consider also the fact that with an orbiting space station, a wide variety of other tasks (which may for the purpose of this discussion be classed as maintenance tasks) must be accomplished. These tasks include assembly operations, servicing and provisioning, inspection, test and calibration, and possibly system modifications and modernizations as system mission requirements change. Assuming that such tasks will be required in space, and the data presented in the previous sections of this report demonstrate the validity of this position, development of a maintenance system capable of performing the necessary functions should be initiated immediately.

There are four general approaches to maintaining the operational capability of space systems:

- (1) Launch a second comparable system into the required orbit upon failure of the primary system.
- (2) Provide an automatic/semi-automatic self-maintenance capability on-board the vehicle.
- (3) Provide a manual maintenance capability onboard the space system.
- (4) Provide a central maintenance facility and a ferry vehicle capability which enables travel to and from the facility and the malfunctioning vehicle.

Approaches (1) and (2) have been classed as automatic maintenance systems; restoration of system function without the on-site activity of a man. Approaches (3) and (4) have been grouped as man-centered maintenance systems due to the fact that man plays a primary role in restoring system function. The following discussion briefly summarizes the general advantages and disadvantages of each approach, discusses some of the development problems which must be solved before utilization of each approach and presents a general maintenance concept framework which includes a requirement for a man-centered maintenance system.

B. AUTOMATIC MAINTENANCE SYSTEMS

1. System Replacement

If a space system has malfunctioned, and lost its scientific or military capability, a direct technique of restoring the required performance would be to launch a second vehicle with similar capability. To approach the problem from this point of view poses a number of critical problems.

In the first place, considering the large number of orbital, lunar and possibly planetary systems planned for the next few decades, it is apparent that the economics of a large number of replacement launches becomes astronomical.

Consider also, however, that a large number of the satellite systems will be manned; a replacement approach denies the possibility of crew recovery, unless a crew escape and reentry system is provided.

Considered from the point of view of time to restoration of system capability, however, the replacement approach offers an advantage over any approach requiring replacement or repair activities; if it is assumed that a standby vehicle for each critical operating space system is maintained in a launch-ready condition. To accomplish this would place an extreme burden on the available launch facilities and drive the operating costs up significantly.

The primary arguments put forth in support of this approach to in-space maintenance are economic. Hughes (120), for example, studied the costs involved in satellite replacement and in-space maintenance and has reached the following conclusions:

- (1) Satellite replacement is the most economical form of maintenance for inexpensive and lightweight satellites. The minimum tradeoff point was estimated to be approximately 10,000 pounds or \$4,000,000.
- (2) Intermediate size satellite systems, ranging up to 70,000 pounds or \$10,000,000 cost are most economically serviced by a maintenance ferry (Earth-to-orbit on call).
- (3) On-board maintenance is most economical only for large, expensive, long mission satellites.

This study, however, included as direct costs of the maintenance system, the costs of developing and operating a complex orbital station including the earth-to-orbit ferry vehicle. With the Military Test Space Station and the Earth Satellite Weapons System, however, these costs cannot be allocated to the maintenance system. The maintenance system cost would consist primarily of the spares payloads launched for maintenance, the extra-vehicular maintenance systems, and the additional ferry

vehicles required because of a shortened system life due to more frequent use. It is to be expected that additional studies of the costs of providing in-space maintenance with large manned systems, of the MISS and ESWs class, will point out clearly the economic disadvantage of the replacement approach to in-space maintenance of large manned orbital systems.

2. Automatic Self-Maintenance

Two distinct approaches to automatic self-maintenance appear reasonable; equipment redundancy, and automatic replacement and repair.

A most effective technique for improving the reliability of a complex space system within the projected engineering state-of-the-art, is through the use of redundant systems. Such a self-maintaining system would include a self-checking computer which would sense a component malfunction, the necessary switching circuitry, and duplicate components or modules as required. While such a design approach can improve the system reliability picture by orders of magnitude, it imposes some critical system problems as the following discussion will indicate.

Assuming a number, n , of series elements required to accomplish some particular function, two approaches to system redundancy are possible; complete standby and parallel modules. (121) The parallel module technique offers considerable advantages over the standby approach. In this method of redundancy, m , elements are packaged in a module containing the required switching circuitry. As one of the m elements fails, a second element switches in and assumes the function. In the simple case where it is assumed that all n elements in a series possess an equal reliability p , the overall reliability is

$$P = \left[1 - (1-p)^m \right]^n$$

P approaches unity when both m and n approach infinity. Thus, a system of extremely high reliability can be developed, regardless of the number, n , elements in series if a large number, m , redundant modules are provided.

It must be noted, however, that it is practical to design redundancy into a space system for primarily electronic subsystems; and even with these systems the redundancy required to meet some operating reliabilities becomes prohibitive from the standpoint of weight and volume requirement. Further, the assumption that the number of parallel modules, m , capable of performing a required function is infinite (or very large) can be reached only if we provide some in-system capability to repair the malfunctioning component. Without such an on-board capability, the restriction in m may well result in system reliability below the required level.

Some of these limitations of the redundancy approach may be alleviated by the development of interchangeable multipurpose, elements which are designed to

perform with the required accuracy, a large number of distinct functions. While some such elements appear possible for closely related functions, the realization of such flexible elements for widely discrepant functions as required in complex space systems, is far from reality and, in fact, may never be possible. Chance Vought and Sperry-Rand (122) have estimated the reliability increases which accrue when interchangeable elements are employed. In these estimates, blocks of 5 interchangeable modules with basic reliabilities of 0.90 and 0.59 were provided with redundancies of 20, 40, 60, 80 and 100 percent, when interchangeability of the spares was present and without it. The data is presented in Figure 58. Investigation of this figure clearly shows, that when interchangeable components are available, and a technique of switching in their function or replacing a malfunctioning component is also available, that reliability is greatly enhanced for a given system weight.

The other approach to automatic self-maintenance which appears technically quite promising is the provision on board a space vehicle of automatic replacement and repair of the malfunctioning system. A remote manipulator system could accomplish some required maintenance tasks without on-site human intervention if the means were provided to diagnose the malfunction, direct the manipulator arms to the malfunctioning component, and control the replacement and repair tasks.

Two different approaches appear possible; complete self-contained and visual feedback to an operator who possesses control over manipulator movement. The self-contained approach would provide in the manipulator system a memory storage capacity of the maintenance tasks possessing the highest probability of occurrence. Thus, once a malfunction is sensed, the diagnosis would be accomplished in a self-checking computer and a maintenance procedure would be read out of storage into the control mechanism controlling manipulator movement. With such a system, the capability of replacing a module or repairing a component would be limited, primarily, by the memory storage capability of the computer. Preliminary estimates of the storage capacities required to accomplish, automatically, the necessary task sequences for maintenance and repair functions indicate that extremely large machine memories are required if the response repertoire of the automatic system is to be capable of handling the wide range of the system malfunctions. Further, the introduction of such a complex computer and memory storage device introduces a source of system malfunction itself which, unless a redundant capability is provided, would result in a maintenance system possessing a fairly low reliability.

Another approach to accomplishing in-space self-contained maintenance is the approach which will be used in the Ranger and Surveyor programs wherein a closed loop television system presents visual information relative to critical aspects of the operating environment. Such a visual feedback system could be used for accomplishing maintenance and repair tasks in an unmanned space system. For example, if a communication link between the satellite and the ground station is continuously available, the closed loop TV system can be activated after a malfunction has been diagnosed and can be used to guide a remote manipulator to the area of the

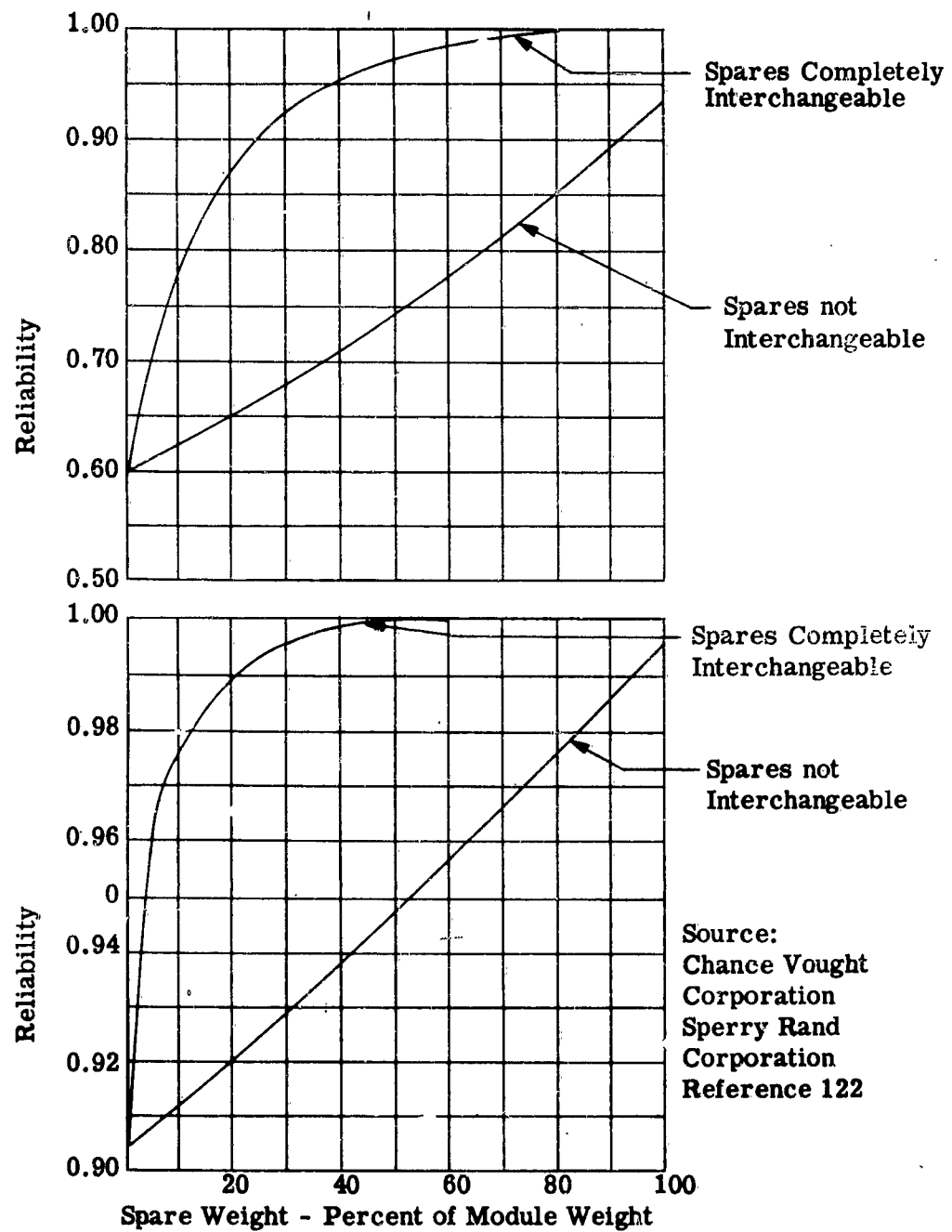


Figure 58. Influence of Spares Interchangability on Reliability

malfunctioning module. If the manipulator system is designed so that its control can be directed by communication link from some distant site to the vehicle, an operator on earth or some satellite orbiting facility can manually guide the manipulator to the module and manually guide the manipulator through the tasks necessary for maintenance. It must be recognized that while terrestrial systems of this type are currently in existence, (123), (124) the tasks which can be accomplished by them are relatively simple. The development of such an automatic system capable of accomplishing a wide variety of complex maintenance and repair tasks is no simple job and would require an extensive development program. The system would have to be designed to operate with very high manual dexterity and be capable of operating in very small work areas and subsequently require complete six-degree-of-freedom control. Once again, the introduction of such a complex system introduces a reliability component in the maintenance system which may well have a fairly low reliability itself.

As indicated above, both the completely self-controlled and the partially manual automatic maintenance systems require considerable development effort before operational systems can be realized. Further, these systems will be fairly heavy in that a rail system on which the manipulators move and the motive forces for movement would by necessity be provided within or upon the space vehicle. In addition, it is predicted that the reliability of these systems will be relatively low and thus may not yield a maintenance system possessing the required reliability.

C. COMPARISON OF AUTOMATIC AND MAN-CENTERED MAINTENANCE SYSTEMS

Before a discussion of man-centered space maintenance is presented it appears appropriate to present some data which evaluate the system reliability expected to occur with automatic systems and systems with manned maintenance intervention. During the NASA sponsored Orbital Launch Operation Studies, (12, 18, 122, 125, 126, 127) one of the principal problems investigated was the effect of manned maintenance on the reliability of systems employed in orbital launch operations.

The rendezvous, docking, and assembly tasks required in the Military Test Space Station (MTSS) and the Earth Satellite Weapons System (ESWS) offer a good medium for discussion. It may be recalled from the discussion of Chapter 3 that considerable accuracy is required of the rendezvous and docking system. Malfunction of any major component will markedly reduce, or even eliminate, the probability of successfully coupling the two segments. Figure 59 presents the estimated probabilities of success for this maneuver with automatic systems and systems wherein man plays a functional role. Mission success estimates are presented for rendezvous maneuvers where one and two rendezvous must be accomplished. Figure 60 presents the individual reliability estimates from which Figure 59 was derived. The estimated reliability for manned rendezvous systems, possessing a repair and replacement capability, was computed by assuming that the man in the loop could repair or replace each component in the rendezvous system. The advantage accrued by an automatic system including a manned maintenance capability is clearly presented in Figure 60b.

The empirical data presented by Starkey (128), comparing manned and unmanned space vehicles, partially verifies the predicted data presented above. In this study, the malfunction reports of a manned system (F8U Crusader Aircraft) and an unmanned system (Regulus I) were compared. The Regulus I data was evaluated in terms of the system losses which could have been precluded if an on-board pilot had been in control of the missile. It was assumed that the pilot was integrated as an on-board functional component with no additional emergency systems at his disposal. The analysis indicated that the F8U suffered only 2 percent of the number of losses suffered by the Regulus I and that approximately 29 percent of the Regulus losses could have been prevented with a trained pilot on-board. The implications of these data appear clear; the use of a man as an on-board functional component of a complex system increases significantly its probability of mission success and enables a significant reduction in system complexity.

These data, while admittedly preliminary, indicate a trend which has considerable implications for in-space maintenance; the major implication being to provide man in the loop wherever and whenever he is able to make a significant system contribution. While not directly related to space maintenance the brief history of the Mercury program is worthy of note. During the initial phases of this program the astronaut's role was considerably more passive than it is at the present time. This was undoubtedly not accidental, however. A few years ago when the Mercury program was being initiated many persons felt that the environments would be too severe and hence, significantly

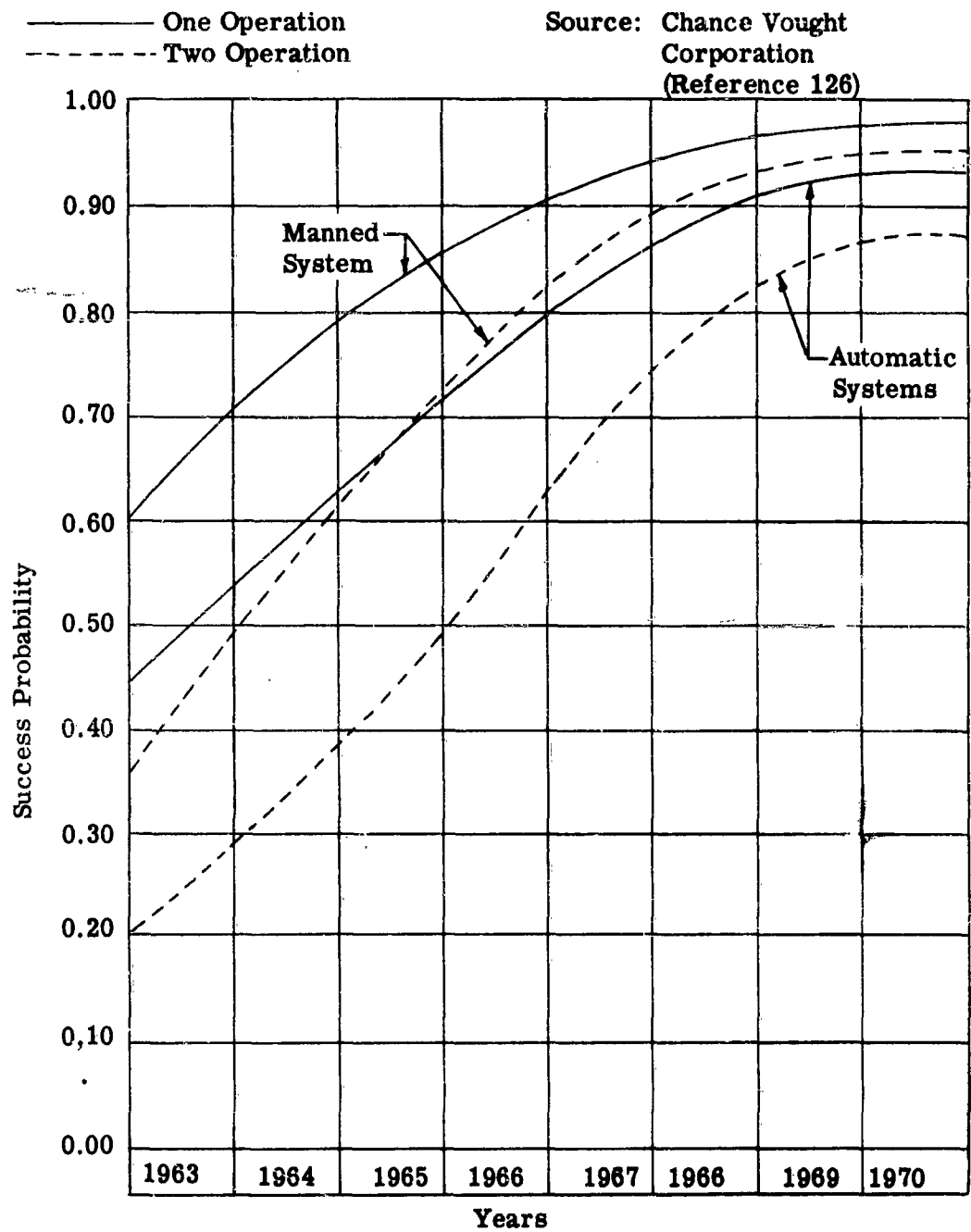
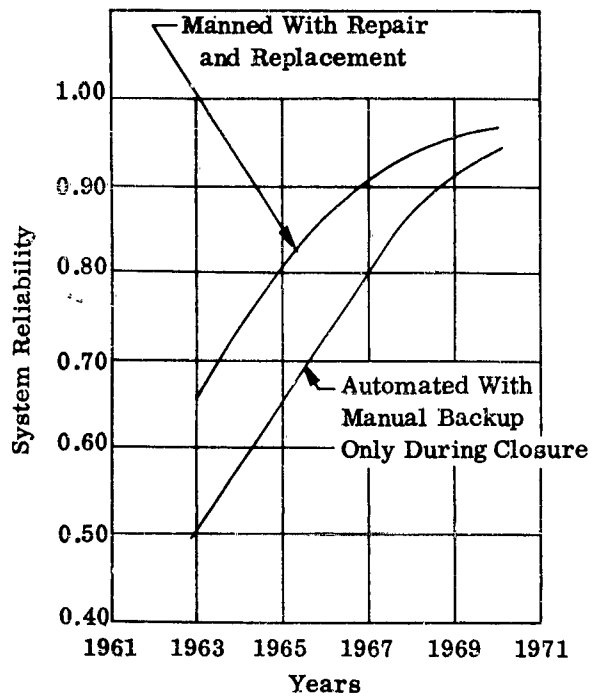
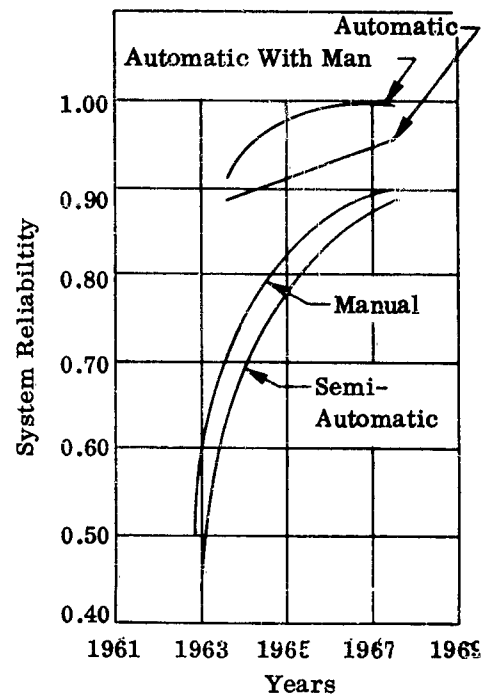


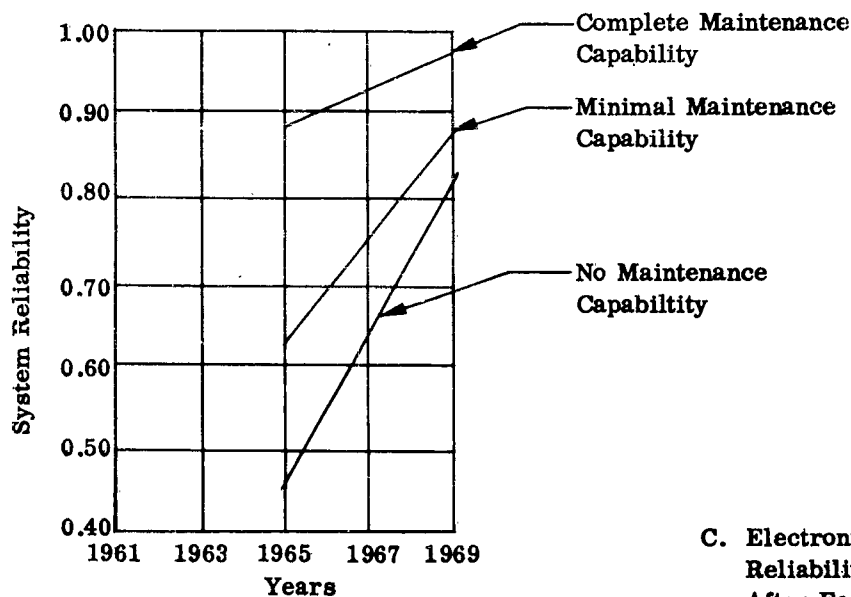
Figure 59. Reliability Comparison Manned - Automatic Systems for Rendezvous, Docking, and Assembly



A. Rendezvous Reliability



B. Retrieved, Docking, and Assembly Reliability



C. Electronic Equipment Reliability 200 Hours After Earth Launch

Figure 60. Reliability Estimates

reduce the capability of the man to function. In addition, the impact of automation was then being felt and the trend was to consider eliminating the man completely, rather than using him where he is most advantageous. The experience with Project Mercury has demonstrated with extreme clarity, however, that when used wisely as a space system component, the astronaut accrues considerable system advantages.

D. MAN-CENTERED IN-SPACE MAINTENANCE

Two distinct approaches to accomplishment of manual in-space maintenance may be posited: a maintenance capability provided by means of a special purpose maintenance vehicle; and a maintenance capability provided by trained personnel on board the space system possessing the necessary support equipments for accomplishing the required tasks.

1. Special Purpose Maintenance Vehicle

The maintenance and logistic vehicle, may be considered as either earth based and launched as required to accomplish in-space maintenance and servicing functions or as orbital based and launched to accomplish the same functions. The decision between these two approaches must be made based upon the economic and temporal factors involved in accomplishing the required maintenance. In the earth based approach, a central maintenance and logistics facility would be located at an optimum launch site which could launch repair vehicles as required. Such a site would eliminate the need for an orbiting facility, but in many cases would, undoubtedly, increase the cost and the time required to initiate the maintenance and servicing tasks over the costs involved in the orbital based approach. Further, it would place a requirement upon the launch facility to have continuously, in a ready status, the number of repair vehicles necessary to accomplish the predicted number of maintenance and servicing missions per unit time. In addition, the servicing vehicles would require the reentry capability for returning the crew to the earth.

It is important to note that once the repair vehicles arrive at the site of the malfunctioning system, the repair vehicle itself must possess the necessary support equipments to accomplish the required maintenance functions or must have on board smaller special purpose systems, which, when disengaged from the ferry vehicle, could accomplish the required tasks.

The space based approach would provide an orbiting central maintenance and logistic facility which could provide the required services by sending a special purpose vehicle from a central location to a malfunctioning satellite as required. In this case, the servicing and repair vehicle could be a much simpler vehicle than the one used in the earth based approach because it does not require the reentry capability. On the other hand, the burden of providing required logistics and trained personnel in space must be considered.

2. On-Board Maintenance Capability

Assume for the moment that a manned space system experiences a malfunction. To the extent that the man on board is not injured, has been trained in maintenance, has available the necessary maintenance handbooks, can diagnose the malfunction, has available the necessary tools and spares, he can attempt the necessary module replacement or repair. In the simplest case, if a system malfunctions and the crew is able to diagnose the nature of the malfunction with some assistance from a diagnostic computer, the man can replace the malfunctioning unit with a spare unit and could therefore provide on-board redundancy which would surpass the redundant capability of almost any automatic self-maintenance system. However, the packaging of space systems is currently not designed to enable in-space maintenance and repair. This point, however, relates with equal validity to any approach, either automatic, semi-automatic or manual, which attempts to enable in-space component replacement. Based upon the analyses conducted in this contract it has been concluded that the larger orbiting space systems should have on board maintenance capability, including a set of basic spares and tools for those systems which have the highest probability of malfunction and which the man can replace or repair from inside the space vehicle. The design criteria which must be borne in mind for the accomplishment of such intravehicular tasks have not as yet been derived. Pigg⁽¹²⁹⁾ and Demaree⁽¹¹⁷⁾ have presented preliminary discussions of the human engineering principles which should be applied. These were expanded in Chapters 4 and 5 of this report, but a complete and valid compilation of the appropriate design criteria must await further detailed study. The problem of system layout and accessibility will be an especially critical problem due to the extremely high density system packaging which is required of space vehicles.

As was clearly indicated in Chapters 3 and 4, there are numerous systems which if maintained in space, will necessitate extravehicular activities on the part of the worker. The accomplishment of such tasks will require the use of some means of protective encapsulation: full pressure suit or maintenance capsule. Considered, for the moment, only from the point of view of protection, it may be concluded that the capsule is superior to the extravehicular suit. Table XXXVIII presents the major environmental parameters which must be protected against and indicates that the micrometeorite penetration problem is the critical one. While the difference in penetration probability between the suit and capsule can be reduced by designing a micrometeorite bumper on the suit^(10, 130), to do so will aggravate the flexibility problems already present in the suit.

From the point of view of task accomplishment both systems offer some critical problems. The full pressure suit with its relatively inflexible gloves does not permit the grasping and handling of small components; further the possibility of penetration when working on maintenance tasks is a critical problem. In addition, the man's reach is limited to his arm's length with this system which may place a severe burden on accessibility design. The use of capsule manipulators, on the other hand, while providing an expanded reach, introduces problems associated

with the use of such devices; increased task time and special design of accessibility areas. The problems involved in the use of manipulators are discussed in detail in Chapter 2.

TABLE XXXVIII
ENVIRONMENTAL PARAMETERS AND THEIR EFFECTS ON
EXTRAVEHICULAR ENCAPSULATION SYSTEMS

ENVIRONMENTAL PARAMETER	RELATIVE EFFECT - SUIT AND CAPSULE
Radiation	No difference between suit and capsule
Vacuum Effects	No significant difference
Electromagnetic	No significant difference
Micrometeorite Penetration	Penetration probability 40 times higher for suit than capsule.

Another important criterion which must be considered in evaluating the comparative desirability of the suit and capsule approaches is the payload weight required for system operation. The capsule weight is approximately two-and one-half times the weight of a pressure suit system (including life support, propulsion, and stabilization and control). In addition, the propellant use rate is significantly greater because of this increased system weight.

A comparative study of a full pressure suit approach and a capsule approach for lunar surface protective systems has recently been conducted by AMF⁽¹³⁾. While acknowledged to be a preliminary evaluation it represents the only attempt, which has been uncovered to date, to conduct a logical analysis of these two extravehicular systems. The evaluation criteria employed were:

- Meteoroid protection
- Heat exchange
- Time of use
- Waste storage/disposal
- Radiation protection
- Food and water supply
- Environmental requirements
- Gravity conditions
- Work condition requirements; tools, mobility
- Safety considerations; warning devices

Vision; glare protection and viewing area
Communications
Illumination
Psychological comfort
Mechanical requirements; weight
Efficiency; input/output

Each criterion was investigated, relative to the capsule and suit and a qualitative decision was made as to which system provided the most protection, least weight, fewer design problems, etc. In total 32 evaluations were made, with the capsule receiving 20 favorable and 12 unfavorable; the suit received 12 favorable and 20 unfavorable. The implication derived from this evaluation was that the capsule should be employed for routine lunar surface travel with the extravehicular suit employed as a backup system. The generalization of this conclusion from the context of lunar surface travel to that of orbital maintenance operations appears legitimate.

E. CONCLUSIONS

The preceding discussion has indicated in a brief manner some of the advantages and disadvantages of automatic and man centered approaches to accomplishing the in-space maintenance. The conclusion which may be drawn from this discussion is that there appears to be no substitute for manual in-space maintenance capability. The capability of the man to function as a means of supplying almost unlimited system redundancy, hence significantly increasing system reliability, and the overall economic and weight savings that appear to result when manual maintenance is utilized instead of completely automatic maintenance indicate clearly the validity of this conclusion. It is recognized that automatic means will be provided to supplement the capabilities of the man whenever they are required, but at the present time it seems a mistake to assume that automatic techniques can completely supplant the man. Basic technological history has shown that whenever automatic systems are introduced into any industrial process, the importance of man in the process becomes more critical even though the total manhours spent on the task might become less by orders of magnitude.

Each of the above described approaches requires the development of the support equipments necessary to accomplish the predicted maintenance and servicing functions. Since the man is present in each of these systems, the fact is recognized that his capabilities should be put into use. Thus, some remote manipulator systems will be present as required to handle servicing functions, and by the same token, the capability should be provided for the man to exit the vehicle and work directly on the malfunctioning system protected by some means of encapsulation, either soft suit or rigid capsule. To date, there are no operational systems which would enable the direct accomplishment of such manual in-space maintenance activities. The following chapters discuss in detail the tool and fastener and remote manipulator system characteristics which are required for in-space maintenance.

CHAPTER 8

TOOLS AND FASTENERS

A. GENERAL

A significant by-product of the task analyses discussed in Chapters 4 and 6 was the identification of the tools needed to accomplish the predominate repair tasks. In view of the unique environment in which the worker and his equipment will be required to function while accomplishing these tasks, an investigation was made of the applicability of terrestrial tools and fasteners to meet the requirements of in-space maintenance. This study involved initially, a literature survey to accumulate background information in the field. Then from a summary of tools, fasteners, and tasks identified in the maintenance analysis a brief experimental program was conducted to isolate potential problem areas and make recommendations for approaches and solutions. The following section discusses this investigation phase.

B. TASK REQUIREMENTS

The large multimanned space stations considered in this contract require orbital assembly functions before operations can begin. Assembly of these stations requires accurate injection of the various elements into parking orbits within the range of shuttle vehicles. Further, it becomes necessary to develop rendezvous techniques and equipment which will permit the manned vehicles to intercept and then assemble the various sections making up the station. It was logical, then, to select as the first task in this study phase the repair of a component which had become damaged during the assembly sequence. The analysis of this maintenance operation identified some thirty-three discrete behavioral steps, an accumulative extravehicular time lapse of 1-1/2 hours for the space worker without including dexterity degradation and three different tools requiring various force applications during use. The fasteners involved were standard AN type screws and nuts requiring a maximum of 12 to 15 inch-pounds of torque for installation/removal.

The Military Test Space Station is composed of a series of cylinders and spheres which are joined to form a large multimanned complex. For this operation the literature suggests use of manned assembly vehicles provided with grapples and manipulators. This assembly task then was the second to be examined and documented. The principal tools and fasteners identified in this task consisted of large (1/2 inch diameter) tension bolts, a ratchet type socket wrench to fit hydraulic flex line nuts and electrical connectors. The torque requirement for the tension bolts was 480-960 inch pounds to be applied by the assembly vehicle manipulator.

Failure reports connected on space vehicle propulsion systems were examined and a failure type selected which appeared predominate in this category of rocket engines. Leakages as a general category and leaks in a particular gasket of the gas generator were then subjected to a detailed task analysis. The logistic vehicle of the ESWS configuration was employed to establish the framework for this analysis. The tools required for removal and replacement of the faulty gasket include diagonal cutting pliers, open-end or socket wrenches, an Allen head wrench, and a screwdriver or special gasket removal tool. The time estimate for this task including suit donning and air lock egress and ingress was one hour and forty minutes.

The repair task analyzed in the Avionic systems discussion was the removal and replacement of a magnetron tube in the tracking transponder of the Military Test Space Station. The tool requirements included a special camlock tool, diagonal cutting pliers, open end wrenches, Phillips head screwdrivers, long nose cutters, and a standard screwdriver. In view of the need for trouble shooting and post installation checkout it was also determined that a small compact test set and welder would be desirable. The estimated elapsed time for this involved operation was 3 hours and 24 minutes including suit donning and air lock egress and ingress times.

One of the most prevalent maintenance tasks encountered during the analysis was that of repairing structural damage incurred by micrometeorite impact. The analysis of task requirements to effect repair of a puncture in the outer wall of the ESWS manned capsule indicated a need for some power tools such as a hole saw, drill, router, or rotary file, a rivet gun and adhesive applying devices. The estimated time required by a space suited worker to accomplish this task was one hour and thirty minutes.

The tasks presented above are representative of those completed during the contract and serve to illustrate the types and sizes of tools and fasteners which are likely to be encountered in a space repair environment. Logically, then, the next step was to investigate the feasibility of accomplishing these tasks while using standard tools and fasteners in a simulated frictionless environment.

C. SPECIAL TOOL REQUIREMENTS

In order to provide a basic working frame work for this study it was assumed, at the outset, that standard state-of-the-art tools and fasteners would be appropriate in the space environment. Were this assumption to be demonstrated as invalid to any degree during the course of the investigation, techniques and designs would be evolved which would provide a solution to the problem. Before establishing this ground rule, a survey and evaluation was made of current approaches to the design of tools for use in space. Reference 131 shows a tool which employs the closed

force squeeze principal to rotate a socket wrench for assembly and disassembly of standard nuts and bolts. A sliding rod near the socket is inserted in a hole near the nut head to provide a reactive path for the torque generated. The torques required for removal and replacement of bolts identified in Chapter 4 varied from a minimum of 20 inch-pounds to a maximum of 250 inch-pounds. The mean grip strength of a representative lot of male subjects has been measured at 108 pounds, without any degradation imposed by suit pressurization or repeated applications (132). Thus a considerable amount of mechanical advantage must be incorporated in the gearing of the device to permit its output to meet the torque requirements indicated. The need for providing an additional hole near all fasteners which may have to be removed, also adds significantly to the problems associated with use of this device. Consider for example a small mass component such as a propellant valve which has been removed from the parent system for disassembly and "O" ring replacement. Employing this tool for removal of valve assembly bolts would present little advantage over conventional means since the tool's zero force transfer feature depends upon a path of resistance through the object itself. In addition, of course, this reaction hole must be sealed in areas of the space vehicles requiring pressurization or other containment.

Three basic tools for corrective maintenance to piping, fittings and flange connections are presented in Reference 133. These consist of closed force devices called a spun fit wrench, a semi-remote version of the spunfit for use in limited access and visibility areas and a nut and bolt tool similar in operation to an extension type socket wrench. Some significant observations regarding use of these tools indicate the requirement for two handed operations in all cases, the gross clearances which must be provided before employment can be realized and the need for initial access to the inaccessible area for attachment of the remote spunfit design. The oversized handles provided on this line of tools were found necessary to permit gripping by a subject wearing a standard aircraft type pressure suit.

Reference 134 evaluated some sixty-five common earth tools for adequacy in the space environment. Pliers, wire cutters and screwdrivers, which were identified in the Task Analysis Chapter 4 of this report, were considered adequate but should be combined with others to effect reductions in numbers required and weight and volume. Open end and ratchet wrenches were also found usable however recommendations were made regarding employment of a more versatile design, a combined tool approach, and reductions in force required to operate as well as in weight.

Some appropriate conclusions were reached during the five degree of freedom simulator program reported in Reference 135.

- (1) Special space tools will not be required.

- (2) Space workers in encapsulation devices will become tired more quickly.
- (3) Auxiliary propulsion is a must for extravehicular activities.
- (4) Rapid adaptability to employing the body as a lever was noticed.

The tasks discussed above required fastening and unfastening torque type fasteners using conventional tools, operating switches and valves and drilling holes all while wearing a conventional aircraft full pressure suit. Each of the task sequences represented simple part tasks; not complete task sequences. A preliminary study of two complete task sequences has recently been reported by Peters, et al (62). The exercise undertaken in this program consisted of removing and replacing a J-2 rocket engine oxidizer by-pass duct and a gas generator sparkplug assembly while wearing a full pressure suit. The conclusions derived from the study were that tool handles should be at least 1-inch in diameter to permit grasping with the gloves employed in the tests; a tool should be provided to simplify handling of small components; and considerable improvements in suit design are required to reduce operator fatigue and discomfort and improve mobility and visibility. A further comment from this program suggests that --- "the method of independently designing general purpose space tools based upon logical requirements, and testing their suitability on hardware, may not yield a complete optimum or effective product." A more realistic approach would consist of an integrated program wherein requirements can be established, designs produced, and test programs conducted for feasibility evaluation.

To verify the conclusions reached during the analysis of the maintenance task information derived during the contract and the conclusions of the studies discussed above, a brief experimental program was undertaken. This work is described in the following sections.

D. LABORATORY EXERCISES

The detailed maintenance task analysis identified the types of terrestrial fasteners, connectors and tools currently employed in aerospace applications. In view of the revolution which would have to take place should the standard threaded torquing fastener be proven impractical for advanced manned space vehicles, it was decided that a program of feasibility testing of current hardware should first be instigated. This program consisted initially of selecting a representative lot of fasteners, connectors, tools and plumbing hardware from the task analysis data, then subjecting these items to a series of laboratory exercises wherein evaluation of their applicability to space usage could be examined.

For comparative purposes, tests were performed under static conditions and also under a simulated single plane frictionless state. Tests consisted of measuring the time elapsed and degree of difficulty encountered in assembly and disassembly of nuts and bolts utilizing standard open-end wrenches, box wrenches, and socket-ratchet

wrenches. AN standard aircraft bolts and stop nuts were used. The bolts were fixed in a vertical position in a vise in a manner such that when the stop nut was tightened, two threads were left exposed on the end of the bolt. Figure 61 shows the apparatus employed. The nut and wrench were placed within reach on the bench. The subjects performing the tasks were instructed as follows upon the signal to begin reach for the nut and wrench; place the nut on the bolt and tighten finger tight; do not spin the nut on; then using the wrench assigned, tighten until the limit is reached (preset with two threads showing); use no more than a 180° stroke; stop with placing the wrench back on the bench; do not try for a speed record, but use a natural pace; and during disassembly, follow the same procedure.

Bolts AN 3, 4, 5, 6, 7, 8, 12 and 14 were used with the stop nuts. The results of the tests are shown in Table XXXIX. The blank portions of this table results from lack of appropriate tools in that category. The tests showed that assembly utilizing the open-end wrench took the longest time, the socket-ratchet wrench took the shortest time, and the box wrench was between the times for the open-end and the socket-ratchet. As the bolt size increased, the time taken for assembly and disassembly increased correspondingly. A leveling off in time is shown for the small size bolts due to the smaller nuts requiring greater dexterity which compensates for the fewer turns required for assembly.

The weightless tests were conducted with the subjects sitting on the Bell Aerosystems Company air bearing platform which provides near frictionless movement in the horizontal plane. The bolts were again mounted vertically in a vise so that torque is applied in the plane of simulated weightlessness and the subjects using the same wrenches performed the tasks of nut and bolt assembly and disassembly. These tests with a single plane of weightlessness can be considered to simulate, in part, the assembly and disassembly tasks which the astronaut would perform in a shirt sleeve environment within the orbiting space station. The subjects were given the same instructions as in the previous tests and in addition were instructed to use handholds and tether and foot bracing as required. Bolts AN 3, 4, 5, 7, 8, 12 and 14 were used. The results of the tests are shown in Table XL. The time required for assembly and disassembly was generally longer than under static conditions.

The AN-3 (3/16 in.) was the only size bolt which enabled assembly and disassembly without a handhold. Due to the small torques required by the AN3 the subjects were able to stabilize themselves to a degree with the tools employed. The box wrench and the ratchet socket wrench were found to be more positive in usage, since they did not have the tendency to slip off as did the open end wrench. Handholds were required on the AN-4 (1/4 in.), AN-5 (5/16 in.), AN-7 (7/16 in.)

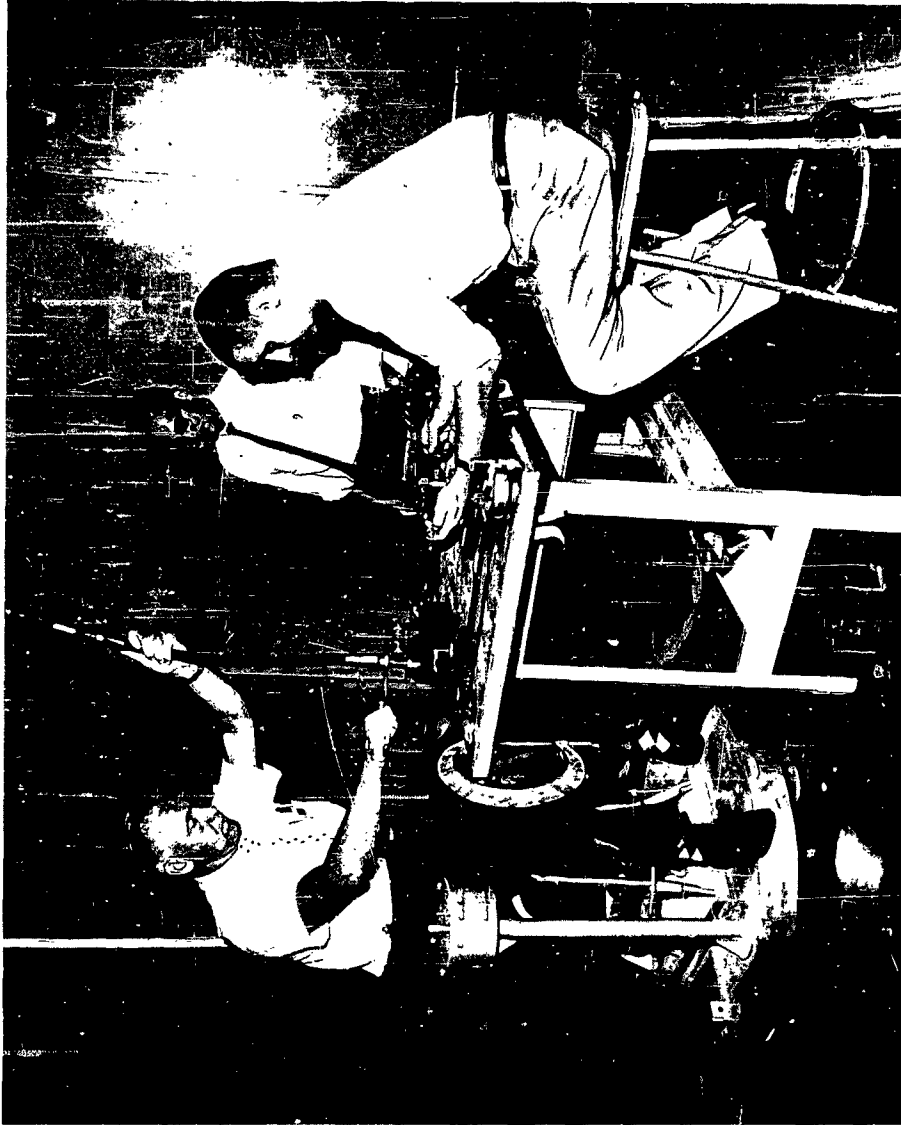


Figure 61. Air Bearing Floor Test Setup

TABLE XXXIX
RESULTS OF TESTS CONDUCTED ON AN STANDARD
AIRCRAFT BOLTS - STATIC CONDITIONS

Bolt Size	Av. time to fasten in seconds			Av. time to unfasten in seconds		
	Open end Wrench	Box Wrench	Socket Ratchet	Open end Wrench	Box Wrench	Socket Ratchet
AN-3	21.0 (4)	19.2 (4)	19.5 (4)	20.0 (4)	20.1 (4)	20.1 (4)
AN-4	24.7 (6)	21.2 (6)	21.3 (6)	21.8 (6)	21.6 (6)	19.8 (6)
AN-5	23.2 (6)	19.7 (6)	19.6 (6)	21.1 (6)	19.3 (6)	19.0 (6)
AN-6	24.4 (3)	23.6 (2)	22.3 (2)	24.7 (3)	24.3 (2)	21.7 (2)
AN-7	28.4 (2)	21.6 (2)	20.1 (2)	31.1 (2)	24.0 (2)	22.0 (2)
AN-8	31.7 (2)	30.9 (2)	24.9 (2)	28.5 (2)	30.0 (2)	24.2 (2)
AN-12	43.0 (3)	-	37.2 (2)	46.1 (3)	-	35.3 (2)
AN-14	49.7 (3)	-	-	41.8 (3)	-	-

() Indicate Number of Trials/Subject.

Two subjects were employed

TABLE XL
RESULTS OF TESTS CONDUCTED ON AN STANDARD
AIRCRAFT BOLTS UTILIZING THE AIR BEARING
PLATFORM - AIRBEARING PLATFORM

Bolt Size	Av. time to fasten in seconds			Av. time to unfasten in seconds		
	Open end Wrench	Box Wrench	Socket Ratchet	Open end Wrench	Box Wrench	Socket Ratchet
AN-3	23.6 (8)	19.1 (4)	19.5 (2)	19.4 (8)	21.0 (4)	19.4 (2)
AN-4	42.8 (3)	33.4 (2)	25.0 (2)	34.1 (3)	31.2 (2)	23.8 (2)
AN-5	30.4 (2)	28.0 (2)	29.2 (2)	27.2 (2)	31.0 (2)	24.1 (2)
AN-7	25.0 (2)	27.1 (2)	24.8 (2)	31.5 (2)	29.6 (2)	23.6 (2)
AN-8	35.1 (2)	34.3 (2)	31.7 (2)	32.2 (2)	33.1 (2)	29.0 (2)
AN-12	47.2 (2)	-	31.6 (2)	42.5 (2)	-	39.7 (2)
AN-14	63.0 (2)	-	-	52.5 (2)	-	-

AN-8 (1/2 in.). A horizontal handhold was found to be the best orientation to react the torque. It was also found that short strokes close to the body were easiest to react with the handhold. For the AN-7 and AN-8, sustained torques to tighten were not possible without a footrest and tether. The subjects, after some experience, found that they could complete the task by applying impulse forces.

In assembly of the larger nuts and bolts, AN-12 and AN-14, a tether with the handhold or foot bracing was necessary. By exerting a push force with the handhold or foot bracing against the tether, it was possible to apply sustained torques in the assembly and disassembly tasks. The tether used in these tests consisted of a woven belt securing the subject to the work bench in a manner similar to a telephone lineman's belt.

Assembly of an AN-16 (1 inch) bolt and a high temperature self-locking nut was accomplished with a crescent wrench utilizing a tether and foot bracing, the task was accomplished relatively easily.

Subsequent tests included assembly and disassembly of an AN-8 bolt and stop nut with an open end wrench using the reaction controls on the air bearing platform in place of a tether or handhold. Assembly was performed in 45.8 seconds and disassembly in 60.2 seconds as compared to 35.1 and 32.2 seconds using a handhold. Difficulty was experienced in coordinating the reaction control forces of the platform with the forces imposed by the assembly and disassembly tasks. Automatic control would simplify the task, however, use of this technique in space would be limited because of the high propellant use rate required. Tethers and handholds would seem to offer many economies in this regard.

Since leakage correction in plumbing networks has been identified as a principle task in space maintenance, assembly of tubing and flexible hose was accomplished with the subjects performing the tasks on the air bearing platform. The fittings were mounted vertically in a vise such that the torque was applied in the plane of simulated weightlessness. It was found that using two standard open end wrenches without tether or handhold, the one inch hydraulic coupling of the tubing was assembled in 21 seconds, and disassembled in 19.5 seconds. A 5/8 inch ID flex line was assembled in the same manner in 15 seconds and disassembled in 15.8 seconds. No difficulty was encountered.

The task of assembly and disassembly of electrical connectors was also examined with the air bearing platform since maintenance of avionic components will involve connector removal and installation. The connectors were mounted vertically in a vise so that during assembly a torque was applied in the plane of simulated weightlessness. No difficulty was found in the assembly and disassembly task. The subjects used a horizontal handhold and manually assembled a typical connector in 21.5 seconds, and disassembled it in 23.5 seconds. For design comparison, a Bendix pigmy type

connector was also included in the exercise. This connector is a bayonet lock-type requiring about one quarter turn for complete assembly. Again, no difficulty in the assembly and disassembly task was experienced but considerable improvement in time was noted. The subjects using a handhold manually assembled the connector in 15.2 seconds and disassembled it in 45. seconds.

Subsequently tests were conducted to determine the maximum amount of torque application possible with the subject in the single plane weightless environment. A Snap-on Tool Corp. Torqometer TQ-50A having an arm of 10.5 in. long was used for the torque tests. This wrench was secured in a vise so that torque applied by the subject while seated on the air bearing platform was in the plane of simulated weightlessness. It was found that with no handhold, tether or other securing devices, the subject could not apply any sustained force in the push or pull direction. However, by applying impulse forces, an instantaneous torque of 500 inch-pounds (force of 47.6 pounds) could be applied.

The application of this impulse, would propel the subject in the direction opposed to the push or pull force applied after which the subject stabilized himself by grasping with the torque wrench or the handhold. The subject found that by first placing himself in motion, then applying an opposing force on the torque wrench, an impulse torque of 400 inch-pounds (force of 38 pounds) could be applied without as great a reacting motion. With one foot braced against the bench, the subject was able to exert an average sustained pull force of 57 pounds. Using only a handhold, an average sustained force of 18 pounds could be applied for a 3-second duration.

E. CONCLUSIONS

The conclusions reached during the experimental portion of this contract, and confirmed by other studies, imply that an astronaut can perform many of the space maintenance tasks within the space station using conventional tools and standard fasteners. Tasks requiring small torques can be accomplished using handholds with the inclusion of a rigid member holding the worker away from the task area (77). Tasks requiring the application of large forces and/or torques can be accomplished using a tether similar to that of a telephone lineman. Care will be necessary in the securing of tools and parts even during use. This was revealed during a test when one subject inadvertently tossed a wrench to his side. In orbit a tool propelled in this manner would cause serious damage to equipment or personnel and be readily lost unless securely tethered to the work area. Since some extravehicular maintenance tasks, such as the repair of a leak in the gas generator of a rocket propulsion system, will require removal of a number of small parts employing several different tools, (14 fasteners and 6 tools in the example cited) consideration should be given to a means of storing these devices to prevent losses from occurring. The use of tools attached by cords to rewinding devices, tool and part clips, pouches in the space suit or a Velcro lined tool box are potential solutions to this problem. (77)

Maintenance tasks consisting of bench type fault diagnosis, repair/rework and tests of components, subassemblies and module assemblies appear feasible using conventional tools. Equipment being worked upon would require being secured to the work bench by items such as clamps, straps, or magnetic chucks. Similarly the individual performing the maintenance tasks must of necessity be securely tied to his work area so he can exert the necessary forces/torques. Conventional tools can be used such as the open end wrenches, box wrenches, sockets, screwdrivers, files, and power tools including electric or pneumatic type. In replacing modules of electronic equipment subassemblies, it appears feasible to use soldering irons in addition to wire wrap and spot welding techniques if atmospheric contamination can be avoided.

CHAPTER 9

REMOTE MANIPULATORS

A. GENERAL

The requirement for extravehicular, in-space, maintenance necessitates direct manual maintenance by a man in an extravehicular suit or maintenance through the use of a manipulator system. The employment of manipulators, and their attendant increases in time for task accomplishment, payload weight and maintenance system complexity obviously should not be utilized unless the tasks demand it. Baker and Crawford (136) in a study of the effect of distance on performance times with a CRL Model 8 Master-Slave Manipulator determined that task times ranged from 6.6 to 9.5 times greater with manipulators than by a direct manual technique. Comparing these time increases to those experienced with pressurized full pressure suits (62), doubling of task time, clearly indicates the low comparative efficiency of manipulator operations. However, the analyses conducted during the course of the contract clearly delineate the need for manipulator systems to augment the capabilities of the maintenance man encapsulated in a full pressure suit. If nuclear propulsion systems are employed, the manipulator system would be utilized for system servicing in a manner analogous to that currently employed in terrestrial nuclear centers. However, even with more conventional orbital space systems, such as the MTSS and ESWS manipulators appear essential for assembly and maintenance tasks because of the expanded reach and mobility they provide the operator. For example, maintenance on the propulsion system of the Saturn IV-b, which is the likely booster to be employed for injection of space station modules, requires a reach of over nine feet to reach some of the plumbing due to its 18 foot diameter. Direct manual maintenance by a man in a full pressure suit is clearly out of the question for such a task.

The employment of manipulators necessitates the use of a vehicle structure to which they are attached and through which the forces and torques are applied. Thus, the manipulators are considered a subsystem of a maintenance (shuttle) vehicle system which is employed for a wide range of orbital tasks. In an analogous manner, manipulators would be employed for extraterrestrial operations but would require different design solutions because of the gravitational fields involved. The utilization of a shuttle vehicle as a maintenance, assembly, materials transport and emergency rescue vehicle places a severe burden on the design of the manipulator subsystem. If such a range of tasks were required of a terrestrial system, the design solution would most likely entail a number of manipulators and grapples each designed for a given class of tasks and each possessing the required force or torque capability and the required number of degrees-of-freedom. The design of manipulators for orbital maintenance vehicles, however, poses numerous additional problems. In the first place the range of tasks must be accomplished by one, or possibly two pairs of

manipulators plus some simpler grapppler mechanisms. Secondly, the manipulators must be controlled from a minimum sized control station with significantly increased feedback requirements due to the task requirements. Thirdly, energy expenditure must be kept to a minimum; indicating the desirability of using operator muscle power output assisted by power-boost, when necessary. Lastly, the space environment places extreme design demands on the manipulators. As a consequence of these problem areas, the development of an efficient manipulator system is probably one of the most difficult and significant problem areas which must be studied relative to space maintenance support systems. The capability of the operator to perform tasks with the manipulators will define the operational capability of the maintenance vehicle and thus determine the extent of in-space maintenance which is feasible.

The state-of-the-art in manipulators has progressed to the point where many manipulator types or classes are available. The various kinds of remote handling jobs, and the various kinds of mechanisms developed for these tasks, provide experience that makes the choice of manipulator characteristics and designs less difficult than in the past. However, extrapolation of past design experience to the selection of mechanisms for an untried task and environment is still a major problem. To be able to apply past experience to in-space remote handling and manipulation systems, many interdependent factors must be considered. It is clear, after the preliminary studies of manipulators conducted on this contract, that none of the existing remote manipulators appear capable of modification short of major redesign in order to be adapted to space use.

The manipulator system is composed of several related elements as illustrated in Figure 62. The system includes an operator distal from a task, a manipulative device, a control unit for dispatching command information, a sensory feedback display and a communication link to span the working distance.

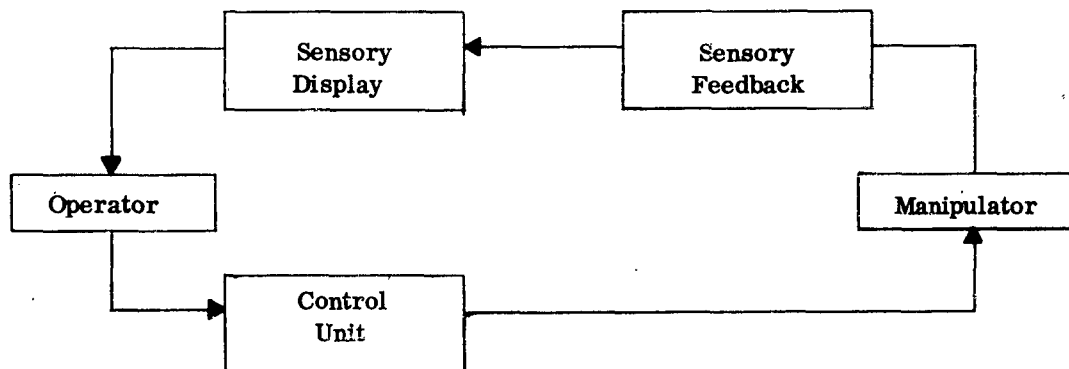


Figure 62. Manipulator System

This conceptual model emphasizes that a manipulator system consists of a loop closed through the operator wherein he serves as a data processor, a decision maker and an issuer of commands. The effectiveness by which he performs tasks depends upon the quality and quantity of sensory feedback and the capabilities of the manipulator. The greatest need in specifying the design characteristics of manipulators is, therefore, detail information on task requirements and those factors which influence operator performance. Four interdependent problem areas are recognized in the selection and design of the manipulator subsystem:

Task Requirements

Mode of Power and Control

Feedback Systems

Environment

A discussion of each of these problems is presented below.

B. TASK REQUIREMENTS

Baker (6) in his Survey of Remote Handling in Space concludes that some five different classes of remote handling tasks will be undertaken:

- (1) Maintenance, servicing and checkout
- (2) Assembly and disassembly
- (3) Experimentation
- (4) Personnel and Materials Transfer
- (5) Emergency operations

A brief discussion of the task requirements of these functional categories is presented below to provide the basis for discussion of the required design characteristics of manipulators.

1. Maintenance, Checkout and Repair

a. Electronics

A black box replacement concept is most appropriate for effecting repairs on externally accessible electronic components. The tracking and rendezvous transponder installation on a typical orbital vehicle was chosen for study of manipulator requirements. Malfunction indications for this system will include; circuit breaker opening in the high voltage tank, erratic beacon responses, or the triggering of low gain indications from the IF Amplifier, Erratic Servo Amplifier Outputs or distorted waveforms on the monitor. If, in the example just cited, the magnetron

cannister requires replacement, the shuttle manipulators must accomplish the following tasks:

Remove access door attachments. These may be standard bolts, screws or small arc Zeus type fasteners or they might be over center toggle latches.

Remove quick disconnect cooling and/or pressurization lines.

Remove electrical connectors.

Remove box attachments.

Remove box and replace with a spare.

Repeat in reverse order steps 1-4.

Adjust new installation for long line effects (Impedance adjustments).

b. Structures

Most rigid structures examined were of the double wall type. Reentry aerodynamic type configurations typically consist of a temperature and impact resistant outer wall constructed of molybdenum or Haynes alloy, an aluminum alloy honeycomb inner wall, an insulation filled space between walls and a tube sheet cooling system for the load carrying structure in areas of high heat flux. Damage to the outer panels of this structure might compromise mission safety during reentry, or a leak in the tube-sheet cooling system could cause structural failure by exceeding the safe operating temperature of the load carrying elements. Accomplishing maintenance with this type of structure would require the manipulator system to remove an outer wall panel. Such a panel has been estimated to consist of 12 in. x 12 in. square panels 1/4 in. thick, held in place by four 3/16 in. diameter screws per panel (approximate torque required equals 20 inch-pounds). Maintenance of the tube-sheet cooling system would require that the maintenance worker be equipped to detect and correct leakages within the sheet itself or from plumbing in the manifold network. Tasks here would require the capability of handling and monitoring a leak detection device (i.e., such as a mass spectrometer designed for vacuum use), fitting and tube replacement and the tightening of leaking tubes, clamps and plumbing fixtures.

Nonrigid structural materials include polymeric cloths with elastomer coatings, thin metallic inflatable structures of Inconel X, René 41, and Upimet 700, as well as numerous plastic materials. These structures are, for the most part, highly susceptible to meteoroid damage and radiation degradation. In addition, some of them require external coating maintenance, on a periodic basis.

c. Propulsion

To effect a repair on liquid propulsion systems, the operator will be required primarily to detect leakages, to repair plumbing tubes and fittings and to remove and replace malfunctioning valves, regulators, and other components. These tasks will require the capability of applying a measurable amount of torque using a wrench or other tool, exerting multi-axis forces during the removal and replacement tasks and grasping and retaining components with varying shapes, sizes and masses.

2. Space Station Assembly

The space station modules to be assembled in orbit are considered to be cylindrical and/or spherical in shape since these configurations provide the structural requirements for internal pressurization and are readily adaptable to the boosters for injection into orbit. If the S-IV stage is utilized for orbit injection, for example, typical space station modules of 10 to 20 feet diameter can be assumed, with cylinder length of approximately 30 feet and module weights of 10,000 to 50,000 pounds.

Assembly of the space station will begin at the time when all modules of the station are in orbit and the assembly vehicles are positioned in orbit within the range of the shuttle vehicle.

Typical assembly tasks have been estimated to be as follows: Each of two maintenance/assembly shuttles proceed to retrieve a module for assembly. Upon arrival, the shuttles secure their attachment arms to standard fittings mounted on the end of the module and position themselves so that the shuttle's thrust will be through the module's center of mass. To enable proper positioning, the standard fittings for shuttle attachment are mounted at the proper locations at each end of the cylindrical module and at one side, and the center of mass at each location is marked. When the shuttle is securely attached to the module, thrust is applied and the module is pushed to the assembly point. Upon reaching the assembly point, the shuttle positions the retrieved module in close proximity and at zero relative velocity to the designated mate. The module is then maneuvered to place the proper end in position for assembly, and gross indexing accomplished. This is performed by the shuttle applying a roll thrust as it is securely attached to the module end. The second shuttle stationed at the mating module provides the information for the indexing maneuver. Then, a closing force (push) is applied until the second shuttle can grasp both modules with its manipulators. This attachment with the manipulator is made such that this second shuttle can act as a pivot. Subsequently, the first shuttle applies the appropriate thrust to the retrieved module, to longitudinally align it with its mate. The first shuttle, then joins the second shuttle, grasping both modules. Now both shuttles apply closing forces to the modules drawing them together. At this time, finer indexing can be accomplished by the shuttles in the course of drawing the modules together. As the modules come in contact, they are secured with temporary snap fasteners. These fasteners are mounted around the periphery of one module and

attach to mating sections on the second module. These mating sections also serve to provide finer indexing as the fastening action is taking place, in the same manner as alignment pins. The fasteners are capable of release by solenoid actuation. At this point in the assembly, the modules are left connected by these temporary snap fasteners to allow adjustment of differences in temperature of the mating sections.

After a period of time, the shuttles return to complete the assembly task. They again take positions at the joint and apply closing forces to both modules, making sure that alignment pins of one module enter the conical leak-in mating hole on the other module and force the modules together. Permanent fastening is then accomplished. This fastening can be made utilizing fasteners such as explosive bolts, or peripheral clamps.

3. Materials Transfer

Shuttle design will incorporate features which will permit transportation of logistic materials from one space vehicle to another. Large manned systems having long orbital life spans such as the Military Test Space Station will require periodic resupply of materials and crews. A ten-man station, for example, requires the following supplies for an assumed 6 week tour of duty:

Oxygen (liquid)	1500 pounds
Nitrogen (liquid)	1000 pounds
Food	750 pounds
Water (for direct consumption)	3500 pounds
Medical supplies	25 pounds
Spare parts, tools, supplies, etc.	50 pounds
Crew rotation	6 men each six weeks

In terms of volume, the water requirement could be contained in a cylindrical tank 4 feet in diameter by 4-1/2 feet long. The food, medical, spare parts and tool supplies could be contained in standardized logistic boxes designed for use in this environment. The fluid tanks and the logistic boxes would be equipped with common fittings for adaptation to the shuttle manipulators. These fittings must be oriented so that spatial translation can be accomplished by directing shuttle thrust through the load mass center, thereby reducing rotational tendencies. A means must be provided for identifying this point and for aligning the attached shuttle such that the thrust vector will pass through the centroid. The logistic tanks and boxes must also be designed for ease of accessibility and mounting within the launched vehicle, since the shuttle will probably be required to first remove a hatch from the primary vehicle, disconnect the container and then remove it from the stowage compartment before translating to the target station.

4. Emergency Assistance

The maintenance shuttle mission capability includes rendering emergency assistance to spacecraft and other shuttles. Vehicles beyond the range of the shuttle, can be assisted in emergencies by having shuttle(s) transported to the emergency site by other spacecraft.

Shuttle assistance can be provided in the form of tools, materials, supplies, and technical personnel from another craft or space station. For example, the shuttle can transport tools and material for repairs; transport supplies such as oxygen, medicine, fuel, replacement parts, or equipment; and function as the transport medium for specialists who must be translated to the emergency site to perform specific tasks.

In addition, the shuttle may be required to transport personnel in emergencies from disabled spacecraft to a space station. An astronaut, in a space suit could be carried by the shuttle's manipulators. In providing assistance to another shuttle, it may be desirable to transport it to the space station. In this case, the rescue shuttle may find it necessary to release the shuttles attachment arms and/or manipulators from its work vehicle. This will require release controls operable from the exterior. The rescue shuttle can then maneuver the disabled shuttle to the space station.

5. Manipulator Operations

It might be said that the most severe restriction on the operation of a manipulator system is met when unplanned tasks are required of the manipulator; the use of manipulators in space operations will certainly involve many such situations. The overall success of the system can be measured, then, by the ability of the manipulators to perform the required tasks as well as the unexpected tasks with minimum time, material breakage and energy expenditure. Thus, it is critical that the potential task requirements of the manipulators be carefully itemized, classified and studied by a systematic analysis of each mission if the manipulator design is to be optimized.

The minimum operations deemed necessary for the manipulator system are: holding, grasping, positioning and actuating. A brief description of each of these operations is presented below.

a. Holding

The holding operation is a stationary one. A small object, e.g., a nut, or a larger object is held with the necessary pressure after having transferred the object from a grasping tool. A holding tool may be employed which changed with the size and shape of the object. The opening and closing of the tool should be possible with the jaws oriented in any direction. The same tool might be used for any shape object

if the jaws stop moving when a pre-set contact pressure is reached. The size of the jaw may depend on the size of the object and the kind of operation involved.

b. Grasping

Grasping is distinct from holding since it involves the operation of closing the jaws around an object which is initially supported by some other means. The grasped object may be transferred into a holding tool, or be deposited. The grasping jaw or tool may be different from the holding one because the latter will be locked for longer time intervals, while the grasping jaw will have to fit into tighter spots. It would be advantageous to have the same tool or jaw for both holding and grasping operations. Tools of any size, such as screw-drivers, cutting and welding instruments, pipe and hose clamps, sealant applicators, and leakage detectors may have to be grasped. Modules of vehicles might be grasped and held, sections of space stations held and maneuvered during assembly operations, in cooperation with the other shuttle manipulators.

c. Positioning

Positioning involves the operation of placing the tool where it is necessary for task accomplishment. Three 3-foot manipulator sections with one hinge-type and three rotary joints appear to be sufficient in length and mobility for covering the predicted work volumes. The employment of such a long arm, however, introduces the problem of manipulator oscillation and difficulty in fine positioning. The number of joints could be reduced to two: one ball and one hinge type, if telescoping tubes were used for the arms. Three feet appears to be reasonable length for the collapsed arm. One ball joint should be arranged at the capsules' wall and at least one at each end of the last section if the three joint design is employed. Telescoping arms should have the hinge at the wall, the ball, or universal joint, at the end of the last arm. Rotations of the arms, telescoping or not, may be realized with rotary pistons or gears.

Positioning must be supplemented by locking the manipulator in position in order that the operations may be performed. The forces applied for locking depend on the task. It is conceivable that, for some operations, steady locking may be necessary, i.e., steadiness with respect to the position between tool and object.

It is anticipated that the tool is placed into the manipulator either while the shuttle is in the mother ship or before the manipulator has been positioned at the work object. The second method of inserting a tool necessitates that the tools be stored outside of the capsule and in such a position that the manipulator can be guided to them and the tools locked on. An alternative solution of this problem is to use a second manipulator for placing and locking a tool on the first manipulator. The two manipulators and the tool box must be so arranged that either manipulator can reach the tool box.

d. Actuating

Actuation of a tool is defined as providing the tool held by the claws, or in place of the claws, with motive power and the possibility of displacement. The tool is in position at the time of its actuation.

A tool is anticipated to serve for any of the following purposes:

- (1) Turn a screw
- (2) Turn a nut
- (3) Lift an object
- (4) Press an object
- (5) Hold an object
- (6) Turn around an object (with help of second manipulator)
- (7) Make a connection: center, bolt on, or screw on, or weld.
- (8) Break a connection: "flame" cut (other steps for breaking a connection are included in one or the other of the operations listed from (1) to (7) above.)

Turning a screw or a nut involves simultaneous rotation around an axis and translation parallel to the same axis. Drive and guidance for these operations may be located in the tool or provided by the manipulator. If the tools drive and guide themselves in these two operations, the manipulators have to position, hold and, very slightly, press. The reaction forces on the shuttle may also be reduced and it may not be necessary to couple the maintenance vehicle to the work piece or to operate reaction controls except during positioning if the tools are capable of self driving and guiding. Ordinarily, the maintenance vehicle would shift around the original center of mass of the capsule manipulator-tool combination during the positioning operation if the capsule were securely fastened to the work piece.

Lifting and pressing may be performed with self-energized tools brought into position and guided by the manipulator. These operations could be accomplished by the same tools or by parts of the tool used to turn the screws or nuts. No external force would be exerted on the manipulator in such a case, if the manipulator were used for positioning the tool, besides guiding it, starting, stopping, and monitoring the operation.

In a different solution, the manipulator may press or lift if necessary, after having placed some guides, mechanical or other, to limit unwanted displacements between the component lifted and its original locus. Reaction forces will have to be contended with in the latter case. The turning around of an object will be performed by simultaneously actuating the motions of two manipulators, each one fitted with a holding claw.

For making connections, the parts must be centered either with the help of built-in guides or by using visual transmission in combination with accurate distance sensors. The latter are important during the last phase of centering. In the case of using welding for final joining, the tool would be held and guided by the manipulator, using either visual cues or impulses as from a curve-follow system. Power would, probably, be supplied from the shuttle through cables running along the manipulator. Claws would hold the welding tool at the end of the manipulator.

Breaking a connection differs from the making of one insofar as it is the reverse and that it may require some pulling. Pulling may be exerted by the manipulator or by separate powered pulling tools, placed into position by the manipulators. The latter may start and stop the sequence of events and help in observing it by moving the viewing optics.

C. MANIPULATOR CONTROL AND ACTUATION

The critical problem involved in the operation of a manipulator system is the accurate direction of the arm to the workpiece and through the necessary motions to effect completion of the required tasks. Since a large percentage of the maintenance tasks, considered for manipulators, require the manipulator to move in and around system structure and plumbing it is essential that they be controlled with extreme precision; the exertion of an undesired force on a system component during the passage of a manipulator to a malfunctioning component could well be as critical as (or more so) the malfunction itself.

Conventional manipulator systems employ either automatic control or manual control of the slave arm. While the employment of automatic control to manipulator arms is now relatively common for industrial-manufacturing purposes, similar employment for in-space maintenance tasks has been studied in only a preliminary manner. The Consolidated Controls Corporation has conducted preliminary design studies of a small remote, memory controlled robot, capable of moving over the external surface of a space vehicle (137) Figure 63 presents an artists concept of this system. They have estimated that such a system, possessing near-human dexterity and strength can be built with a weight of less than 200 pounds. The addition of a memory storage capacity of ten typical maintenance task sequences would result in an additional 30 pounds. The robot could be remotely propelled on a vehicle surface on a monorail system. The employment of a monorail system, however, severely limits the reach capability of the manipulators and results in severe space vehicle packaging problems. In addition the wide range of maintenance tasks which must be accomplished, once in-space maintenance is accepted, dictates that the memory storage capability must be far greater than 10 tasks; and also the variations involved in maintenance tasks must be accounted for. Based upon the data generated during the maintenance analyses conducted on this contract it appears that such problems would negate the use of such a system. While the concept of a memory controlled manipulator system for space maintenance is intriguing, the arguments forwarded in Chapter 7 in favor of man-centered maintenance systems over automated systems indicate the limitations of such automated systems.

The typical approach to manual manipulator control employs the use of spatial correspondence where movement of the master unit, through arm motions of the operator, results in correlated movements of the slave arm. This approach has guided manipulator designers because of the naturalness of the control movements employed and the attendant reduction in operator training required. While appropriate for terrestrial systems its direct application to space systems will result in two significant design problems: (1) the work volume in typical maintenance capsules is severely restricted and (2) the employment of a mechanical master arm results in a considerable system weight. Both of these limitations indicate clearly that the design of in-space manipulator systems must consider the employment of simpler control mechanisms such as the control sticks, toggle switches, or the like, employed in

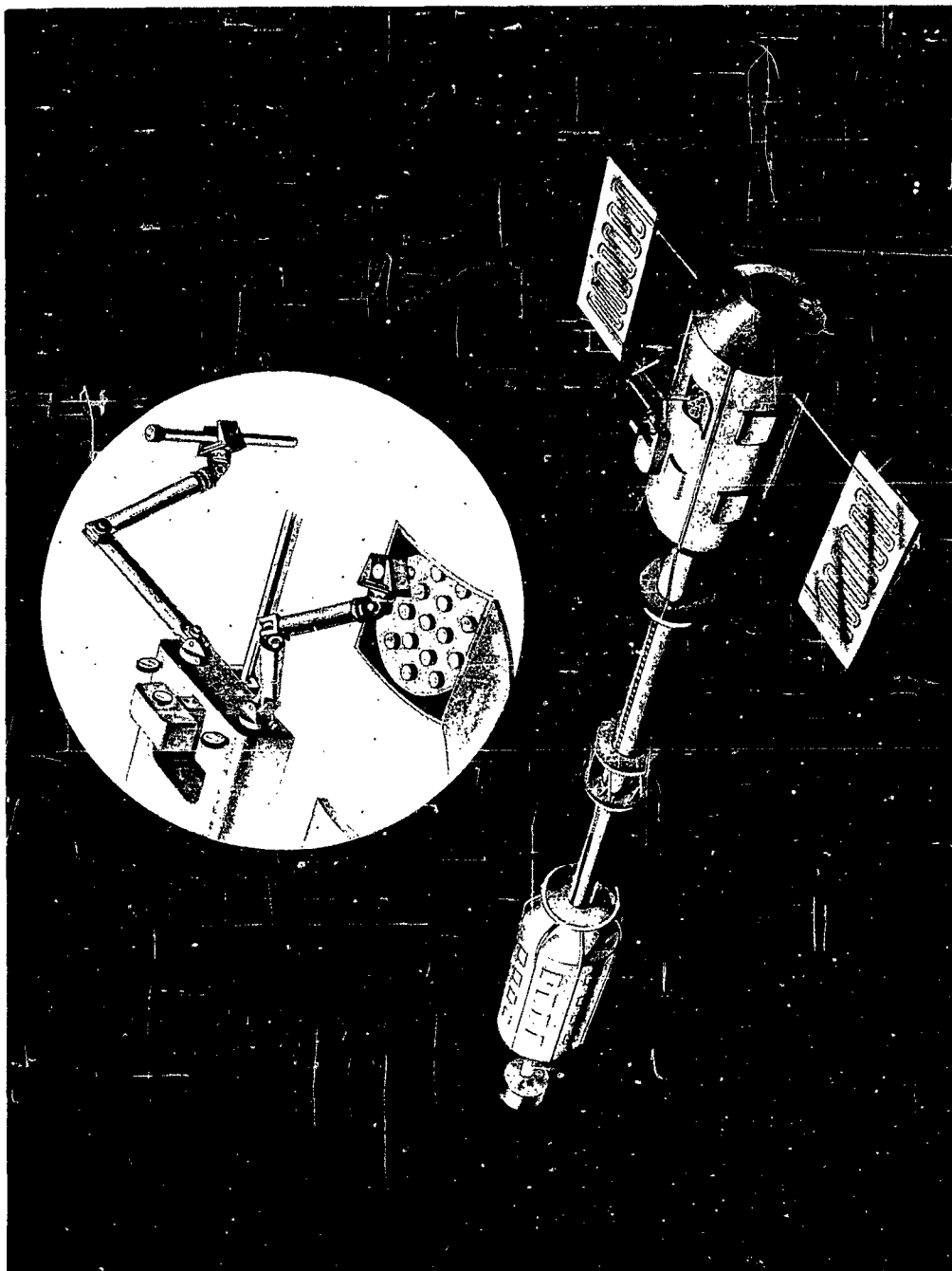


Figure 63. Memory Controlled Manipulator System (Man-Friday)
Courtesy Consolidated Controls Corporation, Bethel, Connecticut

rectilinear arms or power driven electro-mechanical systems. Such control mechanisms, however, significantly increase the times required for completion of maintenance tasks. Marjon ⁽¹³⁸⁾ estimates that a power driven manipulator requires approximately 5 to 10 times the time required by a master-slave manipulator to complete typical manipulator tasks. It is important that detail tradeoff studies of these two classes of manipulator control systems be conducted to determine the optimum configuration for in-space ease

Analysis of the employment of manipulators, for in-space maintenance, has indicated that the system should possess the following general design characteristics:

- (1) The manipulator controls should allow the operator the greatest possible use of natural responses but the employment of simple control sticks, toggle switches, etc. should be considered, where feasible.
- (2) Extensive training periods should not be required to achieve proficiency in use of the manipulator.
- (3) The controls should be designed to minimize operator fatigue during all phases of the mission, including the situation where a full pressure suit is pressurized.
- (4) All controls should return to neutral when the operators hand is removed. This would result in the manipulator arms remaining locked in their last position.

Actuation

Of the basic power transmission, or manipulator actuation, systems available for use with terrestrial systems only two appear to offer the capability of employment in the space environment; electrical and mechanical driven by the muscular output of the operator. Baker ⁽⁶⁾ concludes that hydraulic, pneumatic and hot gas systems should not be employed because of the problems associated with sealing these systems. Electrical systems, while offering a minimum of working parts require substantial electrical power to perform the required maintenance functions. Since such power is at a premium it appears that an all-electrical system is not desirable. Based upon the maintenance analyses conducted during this contract, it appears that the magnitude of the forces required to accomplish a major portion of the manipulative tasks can be supplied by the operator as a direct muscular output, coupled to manipulator arms by a mechanical drive system. Such a system offers the obvious advantage of significantly reducing electrical power requirements and the likely advantage of system weight reduction.

Fundamental energy expenditure rates can be established for accomplishing work based upon the task requirements. Resolution of the amounts of energy and time cycles required will permit the analysis and definition of the mode of power and control. Where the energy output requirements appear to exceed the normal capability of a

human operator, provisions for power boost should be recommended. As currently envisioned, gross mechanical handling problems, such as the assembly of boosters and space stations, shall require auxiliary power to supplement the available human output. On the other hand, the operator's manual effort appears to suffice for the fine manipulation required in repair tasks. If particular situations demand that the operator perform several tasks simultaneously (e.g., manipulate and maneuver the shuttle), then power and control must be engineered to provide such capability.

D. MANIPULATOR FEEDBACK

As indicated previously we may consider the manipulator system as a mechanism which operates from command data and yields sensory data to an operator who closes the loop as a data-processor, and issuer of commands. The effectiveness of the system is dependent upon the completeness of the feedback data. Thus the successful operation of the manipulation system is highly dependent upon the effectiveness of the man-machine relationship.

The sensory feedback systems employed with manipulators, therefore, may be defined as those media which present to the operator the necessary operational cues of the activities of the manipulators. The following sensory inputs can provide these cues; vision, auditory coded data, tactile or force, kinesthetic. Visual and force feedback data are especially critical for the effective use of a manipulator system as the manipulator must first be directed to the proper position, with respect to a work object, and must next be attached to the work object by a force equal to that required for accomplishment of the desired task.

1. Visual Feedback

Direct visual observation of the manipulator task should be provided, if at all possible, including views of both the manipulator hand and the entire manipulator mechanism. However, in many of the predicted maintenance and assembly tasks, required of a manipulator system, direct visual access is not possible and thus indirect viewing methods must be employed. With any of these systems, depth perception is desired if the manipulator is to accomplish the tasks with the desired facility. The following viewing systems are considered applicable.

a. Television

Two or three dimensional television can be applied to provide visual feedback; two dimensional TV, however, does not present sufficient data for accurate judgment of distance. Stereo television can be applied by viewing from two directions (139) or by utilizing stereo attachments to a monocular TV system (140). These systems offer severe disadvantages for a space system, however, due to high power consumption, less reliability and ruggedness than optical systems and sizes which do not always permit access to small, inaccessible work areas.

b. Binocular Periscopes

These devices may be of considerable help in observing areas not within direct line-of-sight through the shuttle window. In addition, when employed with properly etched reticles, they may be used to provide quick and relatively precise judgments of height and distance. They offer the disadvantage, however, of visual fatigue and manipulator control error with image misalignment and inability of the objective lens to be carried into inaccessible areas for viewing.

c. Fiber Optics

A fiberscope is essentially a flexible bundle of fine optical glass filaments with an objective lens at one end, and an eyepiece at the other. The quality of image transmitted by these devices has recently been greatly improved. It is now possible to produce a fiber bundle approximately 1/4 inch square which contains 250,000 image elements. This provides a picture which is comparable in quality to a television image. A fiberscope attached to the manipulator arm could be carried into inaccessible areas or be used for close-up viewing. The image may be viewed through an eyepiece or presented on a ground glass screen. It appears possible to use two separate fiberscopes to obtain binocular vision. Fiber optic systems, however, are heavy and henceforth before a decision can be made as to their employment with the manipulator system, trade-off studies with the other visual feedback sensors must be made.

2. Force Feedback

The successful employment of a manipulator system requires the provision of information relative to the forces generated by the grapplers and through the articulated joints. Without such information, the control of forces or torques is impossible and damage may readily result to both the manipulator mechanism and the work object.

The forces generated by the grapplers (or possibly the articulated joints) can be presented to the operator by the tactile sense, through the physical contact of the hand with the control mechanism, or by visual or auditory displays. Likewise, the kinesthetic and kinematic forces can be sensed by the forces required for movements of the arm or through visual or auditory displays. It is important to note, however, that the employment of a full pressure suit, which may be pressurized during maintenance, assembly, and rescue operations, severely limits the use of direct tactile and kinesthetic feedback and may possibly necessitate the employment of indirect feedback systems.

The feedback systems to be employed in the space maintenance vehicle must be of minimum size to conserve weight and power, but must not be so restricted as to jeopardize the safety of the crew or the success of the mission. Trade-offs exist in the modes feedback information. With little or no kinesthetic or force feedback visual feedback must be very good. Conversely, the stringent requirement for a high quality visual display would appear to be lessened through the provision of "feel" feedback.

E. ENVIRONMENTAL CONSIDERATIONS

The most critical question that arises when subjecting mechanisms to the rigors of a space environment pertains to reliability. While the reliability of remote manipulator

mechanisms operating in normal atmospheres are predictable, the operation of such systems in the space environment poses many questions. This environment will introduce several characteristics which impose design problems. Some of these include (1) the lack of atmosphere, (2) wide temperature ranges, (3) heat dissipation, and (4) electrical phenomena.

Answers to specific questions involving bearings and seals are currently being studied by a number of companies and government laboratories.. Furthermore, advances in technology will be available from current space programs to supply some critical knowledge not yet acquired. Some of the major design techniques which appear promising, for reducing the probability of the manipulator system malfunctioning because of environmental effects, are presented in the following discussion.

1. Hard Vacuum Compensation (Cold Welding)

A major problem that may have to be overcome in the design of the grapplers and manipulators is that of cold welding. The question of cold welding of degassed surfaces making contact in high vacuum is currently under study by a number of research organizations. Some hypotheses have been advanced as to the conditions which might favor cold welding, to wit:

- (1) The surfaces are clean and geometrically matching
- (2) The metals in contact form solid solution
- (3) High contact pressure
- (4) Long contact time
- (5) High vacuum

The last condition might be considered to be redundant because any fresh surface would adsorb gases if exposed to them, and cease being clean.

The danger of cold welding can be reduced by:

- (1) Coating one of the surfaces with a lubricant such as molybdenum disulfide, a low-temperature plastic, or a silicone grease.
- (2) Choose a metal for one of the contact surfaces which will not go into solution with the other, or which forms a permanent, low vapor pressure oxide.
- (3) Serrate one of the surfaces so that the adhesive force be relatively low.
- (4) Do not exceed a safe contact time.

The safeguards against cold welding, if necessary at all, have some consequences for the design of the grapplers and of the tools used and the work pieces handled. If the separating coating were a lubricant, as is the case with molybdenum disulfide, then precautions must be made that tool, grappler and work pieces not slip during operation to such a degree as to impair performance. Shaping of tool and work pieces can assure satisfactory operation. Choice of matching metals may be attempted and if successful, coatings can be dispensed with. The force necessary to perform the task depends on the task and on the time available; the specific pressure between the contact surfaces depends on the serrations. Reduction of the force can be attempted, if necessary. Reduction of contact time is conceivable either by interrupting work or by exchanging the grappler.

2. Techniques of Operating Mechanisms in the Space Environment

a. Bearings

Ball and sleeve bearings can be operated satisfactorily for limited time periods in a hard vacuum with certain available lubricants. Solid films and gold plating show promise for extending service life. Some sealing techniques applied to bearings have been shown beneficial in confining the lubricants under adverse conditions of temperature.

b. Seals

(1) Static

Satisfactory techniques for spatial environment include brazing and welding, certain elastomer-rings and metal-to-metal contact.

(2) Dynamic

Satisfactory techniques include bellows, some elastomers and certain relatively compatible materials such as Teflon and steel.

c. Electrical Components

The present consideration is to seal electrical components where possible with a safe partial pressure to provide desired thermal characteristics, minimize spark and glow, and prevent deterioration of insulations.

d. Transmission of Mechanical Power

There are numerous ways of transmitting mechanical power across or through a material barrier. Some examples are as follows:

- (1) Oscillating metal bellows
- (2) Magnetic coupling
- (3) Sealed torsion tube containing a coaxial shaft to provide oscillating or ratcheting capabilities.
- (4) Sealed helical tube with a coaxial rod to provide translational motion.
- (5) Harmonic flexing gear drive.

To provide ease in manipulator arm maintainability a feature such as is presented by Hughes (6) consisting of an interchangeable, completely self contained arm should be considered. With such a system, if a manipulator arm malfunctions, it could be disconnected and replaced.

F. MANIPULATOR TOOLS AND ACCESSORIES

Accomplishment of the diverse tasks required of a manipulator system will require the capability to operate, often precisely, a large variety of tools and other devices. Most of the presently employed fasteners require the application, initially, of an axial contact force then successive short arc torques through a tool such as a screw driver or a wrench. Once this type fastener is removed, it must be retained or stowed until reinstallation. Replacement of the bolt or screw presents, in addition to the force-torque requirement the need for precise thread line-up prior to application of the turning function. If the manipulator terminus could simulate, in all respects, the human features which it replaces, then standard wrenches and screwdrivers could be employed. Since this may not be feasible within the design envelopes of weight, cost and complexity, it becomes desirable to examine alternate solutions. For example, by replacing the wrench/screwdriver with a drill motor having various end attachments, some of the complex torqueing and holding motions of fastener removal could be reduced or eliminated. Alignment of the fastener for reinstallation could be simplified by incorporating a mechanical two-dimensional leveling angle or fitting into the design of the drill motor. Design of the drill motor chuck might consider use of the "push-to-insert" principle currently used in some hand tools having removable attachments.

In view of the multitude of attachments available for the common terrestrial electric drill, it is not inconceivable that a Universal Space Tool and Accessory Kit might be developed. This tool would consist of a power pack which might be attached to the manipulator maintenance vehicle or to the job itself. A flexible shaft (perhaps telescoping) would connect the power pack to the head wherein the required tool would be mounted. These tools could accomplish not only torqueing operation, but with suitable attachments, also cutting, sawing, and hammering tasks.

Empirical data have shown that the task of leak detection and correction will be encountered frequently during space operations. Plumbing fittings, hydraulic,

pneumatic and liquid propulsion system components are highly susceptible to the form of maintenance briefly described above. Corrective action currently employed requires replacement of faulty gaskets, seals or "O" rings or retorquing of tube nuts to an established design value. Plumbing systems having stringent leak requirements may employ solid brazed or welded fittings. Some joints are necessary, however, for assembly purposes. The manipulator end fitting (hands), in this instance, must be equipped to apply a measurable torque to a tubing nut without damaging or overstressing the joint. Handling and probing with a leak detection device for sensing and isolating a faulty area will be an additional requirement.

The preceding discussion has pointed out some gross manipulator tool requirements and possible solutions. It is deemed important that investigations along similar lines be continued in order to develop realistic manipulator-tool interfaces and design criteria.

CHAPTER 10

SUMMARY AND CONCLUSIONS

A. SUMMARY

The space systems analyzed during this contract represent a broad spectrum of future manned vehicle concepts progressing from the simplest system; an anthropometric space suit, to a complex multimanned earth orbit space station. Between these extremes are the earth-to-orbit vehicle necessary for crew rotation and resupply of orbital complex and the one man space shuttle employed for assembly of the larger satellites.

The missions and orbital parameters of these space systems were established and estimates made of the subsystems which must be provided to effect fulfillment of the mission objectives. Each of the subsystems was analyzed for malfunction trends and space maintenance requirements were established. Pertinent maintenance requirements were selected and analyzed for time significance, tool and fastener requirements, and task requirements by employing a time line task analysis technique. The maintenance tasks predicted for these space vehicles involved both internal vehicle and extra-vehicular activities. Thus it was necessary to investigate the environments which would be encountered by the maintenance worker and to establish the restrictions and limitations which they would impose. These environmental studies, in conjunction with a human capabilities analysis, helped to formulate concepts appropriate to the space maintenance spectrum. Finally, knowing the requirements generated by the maintenance task analysis, the restrictions imposed on human capabilities by various forms of encapsulation, and the hazards and limitations of the environment, it became possible to examine the feasibility of employing standard terrestrial tools, fasteners and maintenance techniques for accomplishing the maintenance activities.

In summary then, this program consisted of the following analytical studies:

- (1) Detailed descriptions of configurations, missions, systems and subsystems making up the Earth Satellite Weapons System, Military Test Space Station, Space Station Logistic Vehicle, Space Shuttle and an extra-vehicular Space Suit.
- (2) Identification of the space maintenance requirements generated by predominate malfunctions within the subsystems of these space vehicles.
- (3) Development of time line task analyses for selected maintenance activities.
- (4) Supporting studies relating to the space environment and Human Capabilities therein.
- (5) Investigations and recommendations regarding applicable maintenance concepts.

- (6) Analytical studies and laboratory exercises assigned to determine feasibility of employing conventional tools and fasteners in the space environment.

B. CONCLUSIONS

The following general conclusions have been derived:

- (1) Direct manned intervention was the most reliable concept for accomplishing in-space maintenance.
- (2) Manned intervention requires extravehicular activities to be fully effective.
- (3) Space suits and capsules should incorporate near terrestrial dexterity provisions to accomplish the maintenance tasks in reasonable times.
- (4) Improvements in protection against environmental hazards such as meteorites and radiation must be incorporated in suits and capsules for long term extravehicular use.
- (5) The use of conventional tools and fasteners is considered feasible with certain modifications and worker restraints.
- (6) Maintenance parameters such as accessibility, serviceability, removeability, identification, and working situations must be enhanced by applying appropriate system design criteria.

Verification and expansion of these concepts and criteria, in follow-on studies, is considered essential if valid maintenance concepts are to be established.

While these conclusions are considered generally valid they must be subjected to detail study in follow-on programs in order to develop appropriate maintenance concepts, procedures and maintenance support systems. The following discussion indicates the direction which these expanded study efforts should take.

1. Space Systems

The space vehicles studied in this contract were conceptual in nature and represented only a portion of the system spectra likely to require space maintenance activities in the future. The data employed in defining the system components for these vehicles were derived for the most part from proposals, periodicals and other related literature sources. At the onset of the program it was obvious that the usefulness of the maintenance requirements studies was directly proportional to the depth of the data available defining the vehicle subsystems and their operating parameters. Obtaining this type of data developed into a major problem area which was never completely resolved.

It is considered vitally important to collect more detailed system data for any further studies in this area. Follow-on studies should attempt to establish a broader and more factual data base including, as much as possible, hardware or near hardware status programs. Some of the current space programs which should be studied in relation to space maintenance are Saint (now called Satellite Inspector Program 621A USAF), Midas 239A (USAF), Gemini (NASA), Apollo (NASA), and Mercury (NASA). These programs have progressed to the point where hard data is now available. In addition study of the Atlas, Titan, Thor, Redstone, Jupiter programs should yield relevant propulsion system malfunction data.

2. Subsystem Analyses

At the conclusion of the maintenance analyses, it was evident that the approach employed had been a logical and feasible one, however, the depth of subsystem analysis was inadequate for the derivation of specific design criteria usable for a broad spectrum of space vehicles. By employing the same rationale and expanding the maintenance analyses and task analyses to include more space vehicle subsystems, a set of specific design criteria and concepts can be produced. Consideration should be given to such vehicle subsystems as hydraulics, pneumatics, crew displays and controls, landing systems, infrared detection systems, data processing systems, and instrumentation. Within these new systems data relative to components required, generic failure rates, failure types, malfunction cues, malfunction verification, and maintenance requirements should be obtained. It will then be feasible to conduct additional task analyses of critical maintenance requirements and establish problem areas affecting their accomplishment. These areas, together with those obtained from an expansion of the tasks analysis phase of the initial contract, will establish the basic groundwork necessary for making specific design recommendations to enhance space maintenance.

The varieties of subsystem types considered during the contract was necessarily limited. In the structural analysis, for example, several conceptual double wall concepts employing different geometric arrangements of hard metallic materials were considered. These configurations were appropriate to the manned space vehicles of this contract. In follow-on studies the vehicles listed above should be investigated; Midas and Saint are unmanned, Apollo, Gemini, and Mercury are state-of-the-art vehicles and hence it is considered likely that different structural concepts will be employed. It is also considered likely that new concepts and approaches will be employed by these vehicles in their other subsystems as well. In addition, different methods must be examined which will permit performance of the maintenance tasks on this class of vehicles.

The propulsion systems analysis shown in Chapter 3 considered the maintenance parameters of major component groups such as valves and thrust chambers. Identification of these groups was obtained from a study which first established the propulsion criteria of the contract vehicles and then allocated, as far as possible, hardware engine configurations which would fulfill these basic requirements. This approach certainly was valid, but at the same time limited, since the detailed hardware information came mostly from Bell Aerosystems Company engines. Thus, it

would seem desirable, during studies that follow to expand the data in this as well as the other subsystem fields. The answer to the question concerning good sources for this type of data leads logically to those programs whose long existence has permitted broad data collection and reduction. Thus, data from projects such as Redstone, Jupiter, Atlas, Titan, and Thor should be considered for inclusion during Phase II. Data such as that included in AF reports MD-60-436 and MD-60-353 which present a breakdown by subsystem of the accumulated failure records in the Titan, Atlas, Jupiter, Thor, and Redstone programs would be valuable

3. Malfunction Diagnosis

Malfunction diagnosis is a broad parameter that ranges from complete reliance on the intellectual capability of the operator/maintenance man to the assignment of all system diagnostic functions to equipment. In either event, one of the basic problems is the determination of the event or events that will be manifest as a result of improper operation and the transmission of this information to the trouble-shooter. Furthermore, this information must be definitive to insure the proper diagnosis. For example, in the maintenance analysis of the propulsion system, the cues to improper operation of the propellant valves include such items as loss of tank pressure, decreased operating time, failure to start, decreased performance, etc. However, at best, with these cues, the crew would only be able to ascertain that the propulsion system was not functioning properly. It is obvious that a complete and detailed inspection of all engine and plumbing components would be impossible. However, it is equally apparent that additional information is required for effective maintenance activity. The problem here is basically one of determining the minimum amount of information that must be made available to provide a rational solution and presenting the information in a form that will facilitate this solution. It seems evident then, that an effort be made to examine the tasks, particularly the malfunction cues, to determine what information must be presented and the ways of displaying this information to the maintenance man/operator.

The mechanical subsystems analyzed during Phase I included structure, propulsion, life support and power supplies. The predominant malfunctions for these subsystems were determined to be micrometeorite punctures in the structure of the vehicles and leakages in seals, 'O' rings or components of the propulsion, life support, and power supply systems. Various methods of rendezvous and docking, and assembly of space systems were also detailed and determined to exhibit component malfunctions due to human errors as a likely mode of failure. In view of the program scope, only cursory attention was given to the status indicators necessary for assessment of malfunction criticality, location, and potential maintenance requirements. Since these status indicators will have a strong influence on future system design it is considered necessary to consider them in detail during the next phase.

The high criticality index applied to system leakages indicates the need for detailed considerations regarding means of providing status monitoring and indication devices for these effects. Prelaunch leak checks of plumbing networks employ an instrument which senses the differences in ionization rate of a standard and a tracer gas which has been injected into the system. A recently developed method employs a "decibel detector" which senses ultrasonic sound pressures produced by leaking fluids or gases. This device does not require a tracer gas and may thus be more applicable to the space maintenance environment. Another type of leak detecting device employs a thermal conductivity sensing detector as a part of a wheatstone bridge circuit for comparing gas free air with gas that is picked up in the area of a possible leak. It is recommended that the feasibility of adapting these devices and other detection and monitoring approaches to space applications and the required design considerations necessary be investigated during follow-on studies.

The results of these activities should provide information relative to:

- (1) The amount of component and subsystem monitoring for all system areas and frequency of status checks.
- (2) Estimate of the amount of on-board and ground monitoring required for the various system and system elements.
- (3) Estimates of system degradation that can be tolerated as a function of mission characteristics.

4. Maintenance Concept Development

The studies discussed thus far have been concerned with the acquisition and generation of data descriptive of space vehicle subsystem configurations, their maintenance requirements and design parameters which will enhance accomplishment of the required activities. In order to be of the most value, the product of such studies should be incorporated into a designers handbook for space maintenance. Before a meaningful approach can be suggested, however, some other aspects of space operations must be taken into consideration. These include studies of logistics implications, maintenance levels for completion of vehicle mission, operational approach for manned intervention of basically unmanned vehicles like Saint and Midas, and concept tradeoffs such as economics versus attainment of mission success and/or technological breakthrough requirements necessary. Consideration of the logistic requirements as established by the maintenance concept would seem to be particularly significant during follow-on studies. For example, assume that from the expanded maintenance analyses it has been determined that meteorite punctures of the Gemini structure exhibits a high probability of compromising mission success. The repair of such meteorite punctures requires on-board supplies of structural materials, fasteners, adhesives and tools as well as the devices necessary to protect and aid the crewman during his extra-vehicular activities. In addition, monitoring and locating instrumentation are required to facilitate rapid detection and localization of the puncture area. The generation of these data will permit analyses of the feasibility of conducting the mission in the vehicle provided, with the necessary logistic backup of spares and tools, and with the capabilities

of the crew and their equipment. In addition, these data will enable the conclusion as to whether the incorporation of these provisions in the existing configuration will degrade the system to an unacceptable degree. The intent here has been to emphasize the lesser recognized aspects of the space maintenance problem and to indicate that these must be integrated with the results of the maintenance analyses studies in order to develop meaningful maintenance concepts.

5. Task Analyses and Maintenance Design Handbooks

The task analyses discussed in Chapter 4 should be extended to define optimum maintenance procedures, particularly with respect to those tasks that involve equipments external to the vehicle. Of particular concern, will be trade-off analyses to determine the extent to which maintenance work should be done externally when there is also the possibility of doing portions of the work internally. Items of particular concern here would include time requirements, quantity and sizes of materials and tools to be handled, dexterity and skills required of the operator, and perceptual requirements of the task, operator fatigue, etc.

As was mentioned previously the usefulness of a study of this sort would appear to be best derived from a presentation of the desired maintainability aspects of space vehicles in the form of a handbook for designers. It is logical to recommend that experienced design engineers be employed as consultants for this activity since they represent the user pool for such a document. Actually there are two types of users (a) those whose interest and responsibility is the entire vehicle (i.e., advanced designers doing configurational and vehicle subsystem installation), and (b) those whose interest and responsibility is to subsystem, component, and part design. Two questions immediately arise: How can data be converted into design criteria in an organized manner? How can designed-in maintainability criteria be presented in an effective manner?

One approach immediately separates the presentation into two basic classifications, manned and unmanned. Each of these is further separated into vehicle classifications, then subsystems, components, and parts. The term vehicle classification, as used here, needs clarification. Space vehicles may be classified as to general function, such as space station, surveillance, ferry, etc., or they may be classified as to size, capacity, number of occupants. In either case, these vehicles have equipment categories that are common, such as propulsion, life support, navigation/guidance, etc. Consider the one-man Mercury capsule. In this vehicle only a small percentage of time can be allotted to maintenance, with none at all being available during critical periods, such as preparation for reentry. Maintenance is also limited by the restrictively small cabin and, without an airlock, exit from the cabin for outside repair is made impossible.

Consider the three-man Apollo Command Module. In this vehicle classification maintenance has a broader scope. The cabin permits mobility of the occupants except for critical periods such as boost and reentry. With three men, more time is available for monitoring and maintenance equipment. The additional space permits storage parts and critical components. The airlock permits outside maintenance. Similar consideration of a space station would show increased capabilities over those briefly discussed above.

A very useful form for presentation of in-flight maintenance design criteria, therefore, would be by the number of vehicle occupants (potential maintenance men). Such classifications might be 1 man, 2 men, 3 men, 3-5 men, more than 5 men. Under each of these classifications, design criteria would be set out for Configurational Design (external to the cabin), Cabin Layout, and Systems Installation.

Having the design requirements in a manner useful to the vehicle designer, the maintenance criteria for subsystems design should be set out against subsystems rather than vehicles. Thus the Propulsion System Designer would find under the heading, "Propulsion Systems", Maintenance Design Criteria applicable to most vehicles.

6. Maintenance Worker Encapsulation

The extravehicular space suit offers the advantage of direct manual maintenance over a rigid nonanthropometric encapsulation approach. However numerous critical limitations are apparent:

- (1) The suit has a significantly higher probability of micrometeorite penetration than the rigid approach. If the suit is hardened by protective materials, it becomes an essentially rigid anthropometric capsule, thereby losing its primary advantage of utilization of the relatively unrestricted movement of limbs for accomplishment of maintenance tasks.
- (2) Many predicted maintenance tasks require the manipulation of malfunctioning components, tools and fasteners at distances greater than that provided by arm reach. To limit maintenance tasks to those within arm's reach would place a severe burden on system design.
- (3) The servicing of nuclear propulsion systems will necessitate the use of remote manipulators, thereby requiring the use of a maintenance capsule.

The optimum approach to extravehicular maintenance would appear to involve the joint employment of suits and maintenance capsules; the suits would be employed for short duration accessible tasks and the capsule for longer duration tasks and tasks requiring extended reach and high force and torque applications.

7. Remote Manipulators

The study has clearly indicated the requirement for remote manipulators for a significant percentage of the required maintenance activities, especially on large space stations where accessibility to malfunctioning components is limited. While it is recognized that remote maintenance should be employed only when direct manual maintenance is not possible, it appears that there is no alternative to the employment of manipulators if space maintenance is to be utilized with a large number of malfunctions.

Detail studies of the following problem areas must be conducted to determine the design criteria for manipulators and their limitations for in-space maintenance.

- (1) Manipulator reach
- (2) Force and torque requirements
- (3) Number of degrees of freedom
- (4) Articulation points to provide access
- (5) Joint rotational capability required
- (6) Number of manipulators and grapple required
- (7) Mode of control and actuation systems
- (8) Man-machine interface
- (9) Feedback requirements and mode of presentation
- (10) Environmental protection problems
- (11) Volume and weight considerations

A survey of published data on manipulator systems has clearly indicated that much of the data developed on terrestrial systems have questionable application to space systems. Further only preliminary research and development efforts are currently underway to investigate the above listed problem areas. Since investigations are dependent upon task requirement data and therefore detail task analysis data must be collected on appropriate maintenance tasks immediately if the development of these systems is to proceed efficiently.

8. Tools and Fasteners

The task analysis of Chapter 4 considered six maintenance activities involving both internal and extravehicular environments. A total of 21 separate tools and pieces of equipment were identified in conjunction with these tasks. The feasibility of employing conventional tools within the context of these tasks was demonstrated on an air bearing floor possessing single plane movement. Due to the limitations in scope, of these studies, however, an expanded effort must be initiated in at least the following areas.

- (1) Expansion of the maintenance task analyses effort and detail specification of the tools and fasteners required for accomplishment of these activities.
- (2) Investigation of possible tool designs which will reduce the numbers, weight and volume requirements. Tool designs which feature multiple applications and weight and volume saving techniques should be given priority. For example, seven different size wrenches, two screw drivers, a lock wire cutting tool and a pair of long nose pliers were required in the six maintenance tasks examined in Chapter 4. By designing a light

weight, positive grip, adjustable tool incorporating cutting and gripping heads and a screw drive head the quantities could be reduced to one or possibly two items. A further savings could be realized by examining the fasteners required and standardizing these to only a few different sizes and types. Other tools which offer weight savings possibilities include the electric drill motor and its attachments, welding torches and rivet guns.

- (3) The laboratory test program conducted during this contract demonstrated the need for investigation of tool restraining and small part storage devices to prevent loss and possible damage to equipment and personnel.
- (4) Leakages of propulsion, life support, and pressurized structural sections was determined to be a predominate failure mode requiring rapid and positive corrective action. The tool provided to locate and measure these leakages must be accurate, portable, quantitative in measurement, light in weight, and operable in the space environment. No known leak detector can meet all of these requirements. Thus it is recommended that investigations be made into the applicability of existing detectors as well as visits to detector manufacturers for discussions of current efforts and potential solutions.
- (5) In view of the limitations imposed by the single plane of motion device employed during this phase, it is recommended that programs be initiated which will employ the USAF C-131B Keplerian Trajectory Aircraft and other multiple freedom simulators. These programs should include investigation and development of restraint devices for personnel and tools, applicability of the multiple and special tool designs and tool and part storage devices.
- (6) Micrometeorite punctures in the structural wall (s) of space vehicles were found to generate another predominate maintenance requirement. Once these punctures are located it becomes mandatory, in many cases, to institute repair activities. The tools and materials identified for these repairs include a drill which was equipped with a hole saw and rotary file attachment as well as twist drills, materials for fabrication of the hole patch, fastening devices such as rivets or screws, and leak sealing materials. The technique and consequently the tools and materials employed for these tasks are obviously oriented around terrestrial approaches which are not necessarily optimized or practical for the space environment. Thus it is recommended that continued studies consider in more detail the various concepts of structural repair, self sealing techniques, light weight squeeze type fastening devices, various adhesives which may perform in the space environment and welding methods which may be applicable.

CHAPTER 11

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<p>aeronautical Systems Division, Dir/Aeromechanics, Flight Accessories Lab, Wright-Patterson AFB, Ohio. Rpt No. ASD-JDR-62-931. STUDY OF SPACE MAINTENANCE TECHNIQUES. Final report, May 63, 290pp. Incl illus., tables, 140 refs.</p> <p>Unclassified Report</p> <p>This report describes the study of space maintenance techniques. Details of the work scope are presented with results of the evaluations and appropriate conclusions. The study was devoted to establishing preliminary concepts for: maintenance techniques, assembly techniques for space stations and other extraterrestrial sites, repair techniques for system components, design criteria for tools and fasteners and criteria for remote manipulators. To accomplish these ends, five manned space systems were studied to identify the operating environ-</p>	<p>1. Space maintenance 2. Space environment 3. Human engineering I. AFSC Project 8170, Task 317008 II. Contract No. AF 33(657)-7938 III. Bell Aerosystems Co., Aerospace/Rockets Div., Buffalo, N. Y. IV. Leonard M. Seale Walter E. Bailey William E. Powe V. Not avail fr QTS VI. In ASDIA collection</p>	<p>aeronautical Systems Division, Dir/Aeromechanics, Flight Accessories Lab, Wright-Patterson AFB, Ohio. Rpt No. ASD-JDR-62-931. STUDY OF SPACE MAINTENANCE TECHNIQUES. Final report, May 63, 290pp. Incl illus., tables, 140 refs.</p> <p>Unclassified Report</p> <p>This report describes the study of space maintenance techniques. Details of the work scope are presented with results of the evaluations and appropriate conclusions. The study was devoted to establishing preliminary concepts for: maintenance techniques, assembly techniques for space stations and other extraterrestrial sites, repair techniques for system components, design criteria for tools and fasteners and criteria for remote manipulators. To accomplish these ends, five manned space systems were studied to identify the operating environ-</p>	<p>1. Space maintenance 2. Space environment 3. Human engineering I. AFSC Project 8170, Task 317008 II. Contract No. AF 33(657)-7938 III. Bell Aerosystems Co., Aerospace/Rockets Div., Buffalo, N. Y. IV. Leonard M. Seale Walter E. Bailey William E. Powe V. Not avail fr QTS VI. In ASDIA collection</p>
<p>ments and identify the maintenance, assembly and repair functions. At the conclusion of these system requirements studies, the capability of the man to accomplish the maintenance activities inside the vehicle and exterior to it was estimated while the man functioned in a "shirtsleeve" environment, in a space suit and in a small maintenance capsule. Simple experimentation, on an air bearing platform was conducted to establish maintenance task times and the feasibility of various fasteners and tools for space maintenance. Preliminary design criteria for tools, fasteners and remote manipulators were established and recommendations for areas requiring additional research were specified.</p>	<p>ments and identify the maintenance, assembly and repair functions. At the conclusion of these system requirements studies, the capability of the man to accomplish the maintenance activities inside the vehicle and exterior to it was estimated while the man functioned in a "shirtsleeve" environment, in a space suit and in a small maintenance capsule. Simple experimentation, on an air bearing platform was conducted to establish maintenance task times and the feasibility of various fasteners and tools for space maintenance. Preliminary design criteria for tools, fasteners and remote manipulators were established and recommendations for areas requiring additional research were specified.</p>	<p>ments and identify the maintenance, assembly and repair functions. At the conclusion of these system requirements studies, the capability of the man to accomplish the maintenance activities inside the vehicle and exterior to it was estimated while the man functioned in a "shirtsleeve" environment, in a space suit and in a small maintenance capsule. Simple experimentation, on an air bearing platform was conducted to establish maintenance task times and the feasibility of various fasteners and tools for space maintenance. Preliminary design criteria for tools, fasteners and remote manipulators were established and recommendations for areas requiring additional research were specified.</p>	<p>ments and identify the maintenance, assembly and repair functions. At the conclusion of these system requirements studies, the capability of the man to accomplish the maintenance activities inside the vehicle and exterior to it was estimated while the man functioned in a "shirtsleeve" environment, in a space suit and in a small maintenance capsule. Simple experimentation, on an air bearing platform was conducted to establish maintenance task times and the feasibility of various fasteners and tools for space maintenance. Preliminary design criteria for tools, fasteners and remote manipulators were established and recommendations for areas requiring additional research were specified.</p>